[redacted]

CCN

82p

X64 12854

Code 2D

DIRECT FLIGHT STUDY USING SATURN C-5 FOR APOLLO PROJECT (U)

NASA CR 55872

CLASSIFICATION CHANGE

TO = UNCLASSIFIED [redacted]

By authority of T.D. No. 74-235

Changed by O. L. Merritt Date 3/22/74

"Available to U.S. Government Agencies and U. S. Government Contractors Only"

VOLUME 1 SUMMARY REPORT

[redacted]

SPACE TECHNOLOGY LABORATORIES, INC.

a subsidiary of Thompson Ramo Wooldridge Inc.

[redacted]

023

DIRECT FLIGHT STUDY
USING SATURN C-5
FOR APOLLO PROJECT

[redacted]

VOLUME I - SUMMARY

(NASA CR --; (STL - 8686-6001-RL000)

Prepared for

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

(NASA Contract Under Contract NASw-516)

0 auth 1 February 1963 92 p 0 refs

2124543

(1) SPACE TECHNOLOGY LABORATORIES, INC.
A Subsidiary of Thompson Ramo Wooldridge, Inc.
One Space Park - Redondo Beach, California (2)

[redacted]

[redacted]

[redacted]

[redacted]

PREFACE

This report presents the results of a study of direct flight for Project Apollo using Saturn C-5. The study has been performed under contract to NASA Headquarters and is the second of two examinations made by STL of the Direct Flight mode.

The present study has examined design criteria for the direct flight mode and has defined the differences between the criteria and subsystem weights used by NAA in their Apollo program and STL in its earlier study. A detailed preliminary design of a two-man direct flight spacecraft has been made. The modifications required to implement a rescue mode requiring unmanned landing and protracted unattended stay on the lunar surface have been defined. The propulsion stage designs required for direct flight have been somewhat refined over the earlier study.

Volume I of this report summarizes the results and conclusions of the study. Volume II presents the details of the analysis and design work.

The active assistance of Dr. M. Alper, the NASA Headquarters study director, is gratefully acknowledged. Mr. Clifford Mercier of North American Aviation, Space and Information Division, was most helpful in providing needed documents and arranging for a useful meeting with NAA personnel.

[redacted]

CONTENTS

		Page
1.0	INTRODUCTION	1
2.0	STUDY CONCLUSIONS	4
3.0	SUMMARY OF RESULTS	6
3.1	Mission Sequence	6
3.2	Spacecraft Sequence	6
3.3	Velocity Increments	10
3.3.1	Velocity Margins	12
3.4	Spacecraft Payload Performance Capability	12
3.5	Command Module and Associated Support Equipment	18
3.6	Payload Margins	22
3.7	Spacecraft Propulsion Stage Design	26
3.7.1	Launch Configuration	26
3.7.2	Lunar Spacecraft Arrangement	29
3.8	Command Module and Associated Equipment Design	36
3.8.1	Command Module Configuration	36
3.8.2	Subsystems	51
3.9	Rescue Mode	69
3.10	Comparison of the STL 3-Man System Design of Reference 1 with the NAA Apollo	72
4.0	REFERENCES	76

[redacted]

[redacted]

ILLUSTRATIONS

Figure		Page
3-1	Mission Profile	7
3-2	Lunar Spacecraft Stages	9
3-3	Lunar Spacecraft Payload Performance (NASA Velocity Increments + 2% Velocity Margin)	16
3-4	Direct Approach Configuration, Saturn C-5	27
3-5	General Arrangement, Spacecraft and Launch Abort System (154-Inch Diameter Command Module, Cryogenic Propellants)	28
3-6	Inboard Profile, Deboost Stage (Cryogenic Propellant)	31
3-7	Inboard Profile--Lunar Landing and Take-off Stages	33
3-8	Pressure Vessel Basic Dimensions	37
3-9	Inboard Profile--138-Inch Diameter Command Module	39
3-10	Crew Arrangements, Two-Man Command Module	41
3-11	Interior Arrangement, Two-Man Command Module	42
3-12	Command Module Crew Stations and Volumes	43
3-13	Basic Dimensions (123-Inch Diameter Command Module)	45
3-14	Inboard Profile--Command Module	46
3-15	Pressure Cabin Crew and Equipment Space	48
3-16	Equipment Arrangement Crew Compartment	50
3-17	Recover System Arrangement for Two and Three-Man Command Modules	52
3-18	Earth Landing Impact Attenuators	53
3-19	Guidance and Navigation Summary Block Diagram	55
3-20	Stabilization and Control System Block Diagram	57
3-21	Communications and Instrumentation Block Diagram	60
3-22	Crew Restraint System	62
3-23	Schematic Diagram of Environmental Control System	63
3-24	Expanded Scale Power Profile	67
3-25	Electrical System Schematic Diagram	68
3-26	Control and Display Arrangement	70
3-27	Command Module and Support Equipment Gross Weight	75

[redacted]

TABLES

		Page
3-1	Mission Sequence	8
3-2	Direct Flight Mission Velocity Increments	11
3-3	Spacecraft Performance Uncertainties - 3σ	13
3-4	Spacecraft Payload Capability	14
3-5	Effect of Command Module Size - 8200-pound Payload	17
3-6	Summary Weight Comparison at C-5 Liftoff	19
3-7	Summary Payload Weight Comparison	20
3-8	Subsystem Weight Comparison	21
3-9	Command Module and Associated Support Equipment Weight Changes Between Liftoff and Re-entry	23
3-10	Payload Margins with 2 Percent Velocity Margin (NASA Velocity Increments - 90,000 pound Injected Weight)	24
3-11	Payload Margins with 10 Percent Velocity Margin (NASA Velocity Increments - 90,000 pound Injected Weight)	25

1.0 INTRODUCTION

The feasibility of manned lunar landing and return using a direct flight mode (no orbital rendezvous) and a Saturn C-5 launch vehicle was examined by STL in the spring of 1962. The study (Reference 1) concluded that direct flight was feasible with large margins if cryogenic propulsion stages were used for the larger velocity increments needed for lunar capture and deboost and for the earth return phase. A command module of 138 inches in diameter was used for the three-man crew. The system was designed for a ten day mission including three days on the lunar surface and operation during both the lunar day and lunar night.

The overall weight of the command module and equipment module described in the initial study was substantially lower than the overall weight quoted for the Apollo system. There were also significant differences in the subsystem weights for the two systems. It was not clear, however, how much of these weight differences could be attributed to deviations from NASA design guidelines nor was there any clear statement of what design criteria had been used by STL. The relatively short time period of the initial study (six weeks) had precluded any careful accounting and reporting of the criteria used.

The present study had,as one of its tasks, a reconciliation of the design criteria and weights used in the original study with those used in the current Apollo. The objective of this task was, primarily, to establish a set of subsystem weights which could then be used in the design of a two-man spacecraft for the direct flight mode. Another objective was to show more explicitly the relationship of the system design criteria used by STL to the subsystem weights which resulted. It was desired, moreover, that all deviations to NASA (and Apollo) guidelines be documented and justified. The justification was to include considerations of weight and performance, mission reliability and crew safety.

The task of comparing the STL command and equipment module design and design criteria to those used by NAA in the Apollo program proved to be quite difficult for several reasons. For one, the Apollo is an evolving system and the published documentation does not completely describe the system and subsystems since these are in various stages of design and development. Certain difficulties were also encountered in comparing subsystems weights as a result of differences in "bookkeeping." Nevertheless, it was possible to account for most of the weight differences and to relate these to specific and definable causes.

STL in the present study deviates in certain fundamental respects from the NASA Apollo guidelines as outlined in the Apollo Work Statement (Reference 2). A large fraction of these deviations results from the selection of design solutions which fundamentally differ from those stipulated by NASA. However, many of the differences between the NAA Apollo design and the STL design appear to be disappearing as a result of the Apollo weight reduction program. In the area of propulsion for the lunar spacecraft, STL continues to advocate the use of cryogenics for the larger velocity increments including lunar return. The propulsion stage designs recommended by STL are based on the use of the Pratt and Whitney RL10A pump-fed engines which are far advanced in development and have shown high reliability.

The system designs which have emerged from the present study bear a close resemblance to those presented in the previous STL study, Reference 1. Advantage has, however, been taken of the additional study effort to simplify the designs and obtain additional weight reductions. This has resulted from a more thorough analysis of requirements and mechanizations.

The principal performance results remain substantially as stated in Reference 1. However, the margin between spacecraft payload weight capability and command and equipment module weight requirements has increased. Part of this increase is due to the more efficient designs and part is due to a correction in the "bookkeeping" procedures.

The command and equipment module weights which have been quoted by STL for the direct flight mode require different interpretations when they are compared to the NAA Apollo on the one hand than when they are used in conjunction with spacecraft performance for margin determinations. This is an important point which was not fully appreciated until just prior to publication of this report. As a consequence the margins are now greater than they were when presented at the final briefing while the differences from Apollo are smaller. This subject is discussed in some detail in Section 3 of this report.

Volume I summarizes the principal results of the study. Descriptions of the system designs are brief and no attempt is made to justify any of the results presented. Volume II of this report presents the details of the design and analysis.

[redacted]

2.0 STUDY CONCLUSIONS

The principle conclusions reached in the study are summarized below.

a) Lunar direct flight is feasible using the Saturn C-5 as a launch vehicle.

b) Based on a 90,000 pound injected weight the payload capability of a lunar spacecraft which uses pump fed cryogenic H_2-O_2 propellants for all of the major velocity increments is 12,452 pounds when STL performance margins are used and 10,458 pounds when the NASA 10 percent velocity margin is used.

c) A 138 inch diameter command module is suitable for housing a crew of three astronauts. The nominal weight of the command module and its associated support equipment for a 10 day mission is 8475 pounds at liftoff. The estimated weight including an STL weight growth contingency is 8850 pounds. The corresponding nominal payload weight to be used in conjunction with lunar spacecraft performance estimates to determine payload weight margins is 7515 pounds.

d) A 123 inch diameter command module is suitable for housing a crew of two astronauts. The nominal weight of the command module and its associated support equipment for an 8 day mission is 7351 pounds at liftoff. The estimated weight including an STL weight growth contingency is 7652 pounds. The corresponding nominal payload weight to be used in conjunction with lunar spacecraft performance estimates to determine payload weight margins is 6410 pounds.

e) The payload capability, based on a 90,000 pound injected weight, of a lunar spacecraft which uses pump fed cryogenic H_2-O_2 propellants for all major velocity increments through lunar main descent and pressure fed storable propellants for lunar landing and return is 8261 pounds when the STL margin is used and 6574 pounds when the NASA 10 percent velocity margin is used.

[redacted]

f) The use of cryogenic propellants for the lunar spacecraft propulsion stages is feasible. The principle problems in the design of the propellant thermal control system result from the earth atmospheric phases prior to and during launch. These are solvable. Boiloff during space operations is quite tolerable even for rescue missions which require a 30 day stay on the lunar surface.

g) The Pratt and Whitney RL10A series engines have been used in the STL "preferred" lunar spacecraft configurations. These engines are in an advanced state of development and show a high reliability.

h) The current uncertainty in the severity of the micrometeoroid hazard is such that at the "optimistic" levels the probability of puncture of a vital part of the lunar vehicle during an 8 day mission is about once in 1000 missions. At the "pessimistic" levels, the probability of puncture is about once every 3 missions. The weight penalty required to reduce the probability of puncture (at the pessimistic levels) to once in 1000 missions is about 615 pounds for the command module, 1500 pounds for the lunar return stage and 4200 pounds for the deboost stage. This degree of uncertainty requires an early resolution.

i) The rescue mission mode can be implemented with the STL system designs with only minor additions to communications and instrumentation equipment. The weight penalty to the system is negligible except in structure. An estimated 650 pound increase is required for micrometeoroid protection to maintain the risk of puncture for a 30 day mission at the same level as for a normal mission.

[redacted]

3.0 SUMMARY OF RESULTS

3.1 MISSION SEQUENCE

12854 A

A Saturn C-5 launch vehicle, capable of injecting 90,000 pounds to the moon, boosts the Apollo direct flight spacecraft into an earth-lunar transfer trajectory. The spacecraft coasts to the moon making the appropriate midcourse corrections and deboosts at pericynthion into a circular orbit. The spacecraft then deboosts out of orbit and lands at the desired lunar site. A day or two later, the spacecraft is launched from the moon, is injected into an earth return trajectory and coasts back to earth making the appropriate midcourse corrections. Upon nearing the earth, the command module separates from the rest of the spacecraft, and commences a guided lifting re-entry. A final parachute descent completes the return of the command module to the vicinity of the desired earth landing site.

The mission considered in the present study takes place over an eight day period (or less) and allows a capability for remaining on the lunar surface for at least 24 hours. In addition, a capability has also been provided for crew support for a three-day recovery period after earth landing.

The mission profile is portrayed by Figure 3-1 and the major mission phases and time of event occurrence are shown in Table 3-1. Conf. AUTHOR

3.2 SPACECRAFT SEQUENCE

The spacecraft is made up of four physically distinct stages (or modules). These are:

Command Module
Lunar Takeoff Stage
Lunar Landing Stage
Deboost Stage

These stages and their combination into vehicles and spacecraft are shown in Figure 3-2.

[redacted]

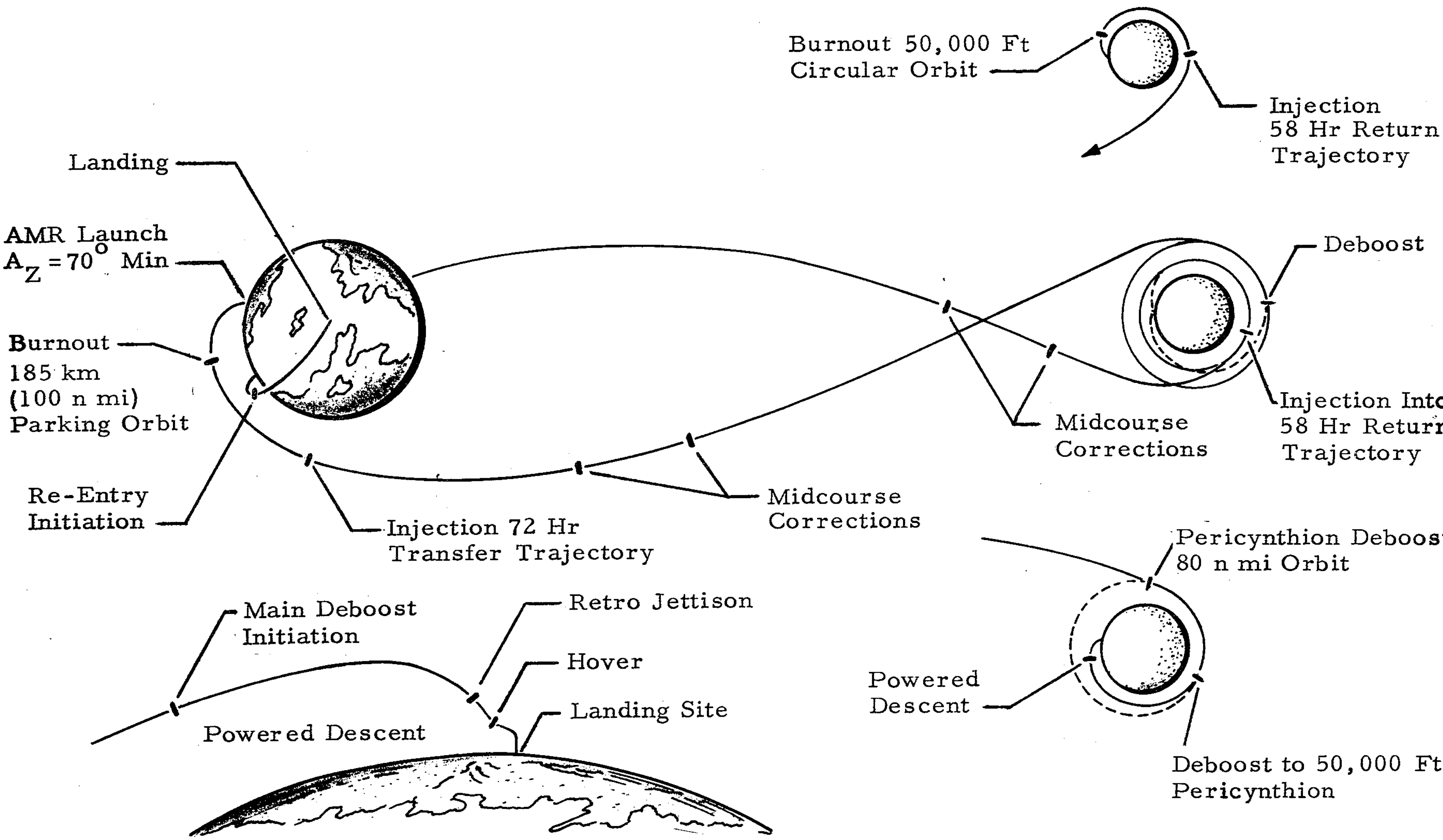


Figure 3-1. Mission Profile

Table 3-1. Mission Sequence

	Phase	Time (hr)	Event
1.	Injection into lunar transit trajectory	2	Inject 90,000 lbs to Moon on 72-hour trajectory
2.	Pericynthion retro	74	Retro at hyperbolic pericynthion to inject into 80 n mi lunar orbit and adjust inclination
3.	Deboost to Hohmann transfer	77	Retro to 50,000 ft pericynthion altitude
4.	Main deboost	78	Retro to 1000 ft, then jettison deboost stage
5.	Final landing	78	Select the landing site, translate, hover and land
6.	Lunar surface (nominal)	78-102	Lunar surface day or night operations - conduct scientific experiments
7.	Lunar takeoff to orbit injection	102	Lunar launch and inject into 50,000 ft altitude orbit
8.	Boost to Earth injection	104	Inject into 58 to 87 hour trajectory for Earth return
9.	Re-entry	191.5	Lunar takeoff stage jettisoned, command module re-enters and lands
10.	Recovery	192-264	Recovery operations

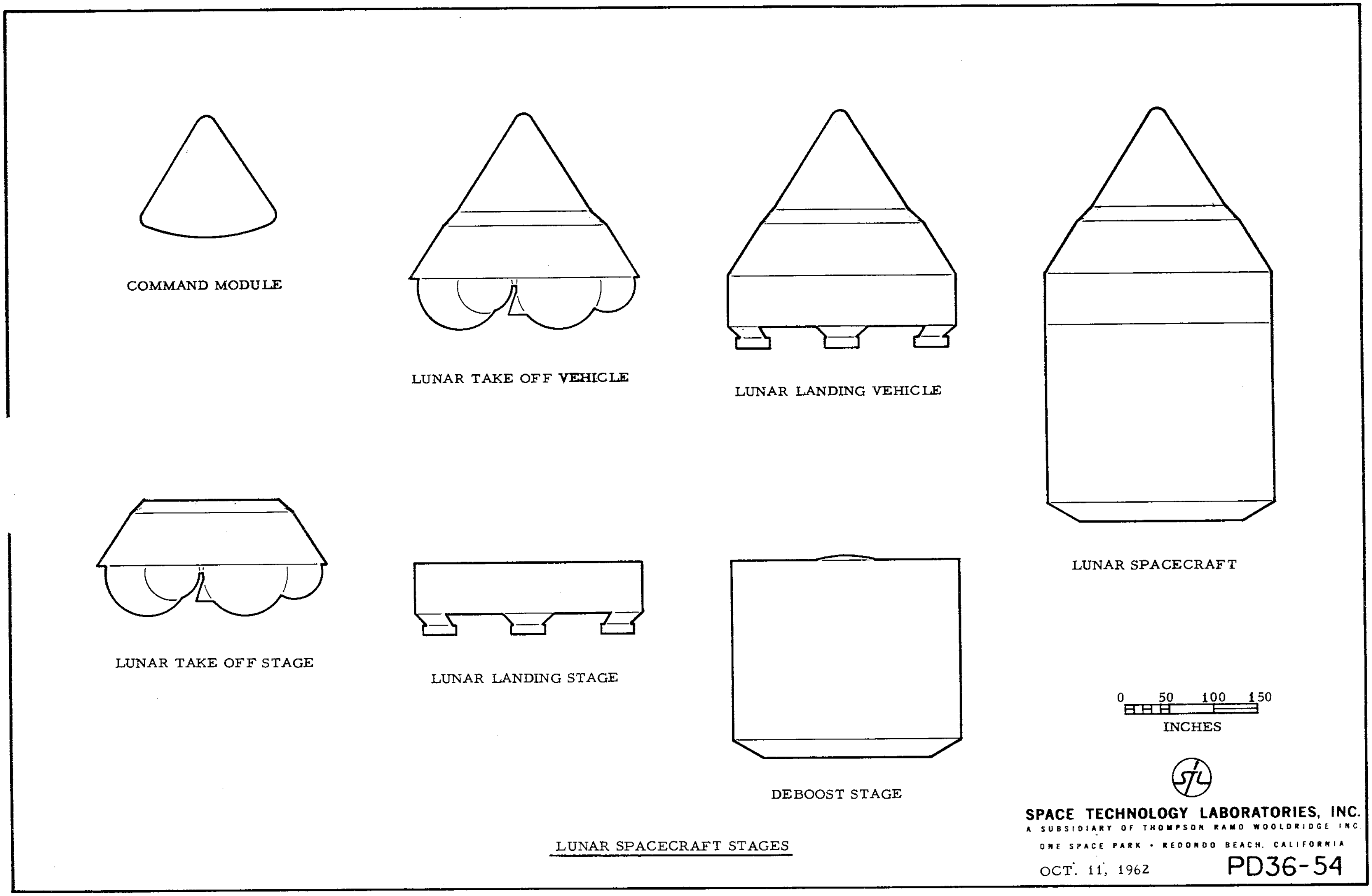


Figure 3-2. Lunar Spacecraft Stages

The complete spacecraft performs the transit to the moon. The deboost stage is jettisoned after the vehicle has been brought to an altitude of 1000 feet above the moon and the lunar landing vehicle completes the descent. For the return trip, the lunar takeoff vehicle is launched off of the lunar landing stage. Just prior to earth re-entry, the lunar takeoff stage is jettisoned and the command module completes the descent maneuvers.

3. 3. VELOCITY INCREMENTS

The mission velocity increments are determined by the trajectory profile and the spacecraft propulsion parameters. In this study, a specific set of velocity increments have been used by STL at the direction of NASA (References 3 through 5). While the increments are in some respects different from values which would be characteristic of the STL spacecraft and propulsion configurations (due to differences in thrust to weight ratio, specific impulse, etc.), the NASA increments have been used in the spacecraft propulsion stage sizing studies and for computing the available payload for the several propulsion options. The increments which are characteristic of the STL configuration have been shown in Table 3-2. The velocity for the lunar descent and takeoff phases are dependent on configuration characteristics and should not be arbitrarily specified. A detailed discussion of the several increments on the table is presented in Volume II.

It is believed that the choice of a return flight time of 58 hours by NASA imposes a payload penalty that far exceeds any apparent benefit derived by completing the mission approximately 5 percent sooner. This payload penalty is 515 pounds for the STL cryogenic configuration when compared with the payload for a 68-hour return. Consequently, the slightly longer return time has been recommended. However, the spacecraft design and its available performance margin has been based on the NASA velocities, including the 58-hour return.

Apparently the NASA velocity value for the Hohmann transfer deboost from lunar orbit shown on the table is an oversight since it is compatible with a deboost from a 100 nautical mile orbit instead of the 80 nautical mile orbit being used.

[redacted]

Table 3-2. Direct Flight Mission Velocity Increments

Mission Phase	Velocity Increments: STL[1] (ft/sec)	NASA[2] (ft/sec)	STL Recommended (ft/sec)
Translunar			
Midcourse	250	300	300
Retro into lunar orbit	3090 (100 n mi)	3130 (80 n mi)	3130 (80 n mi)
Simultaneous plane change	91	100 (6^{o})	100 (6^{o})
Lunar Orbit to Lunar Surface			
Hohmann transfer to a 50,000 ft pericynthion altitude	122	123	97
Descent to 1000 ft altitude	5404	5961	5780[3]
Hover, translation and landing	1045	700	700
Lunar Launch			
Launch to circular orbit	5954 (100,000 ft)	5885 (50,000 ft)	5785[3] - 5800[4] (50,000 ft)
Transearth			
Transearch injection	3194 (68 hr)	3592 (58 to 60 hr)	3194 (68 hr)
Midcourse	250	300	300

Note: The NASA values were used by STL during this study for stage sizing and payload determination.

(1) Used in Ref. 1
(2) Reference 2
(3) Characteristic of STL cryogenic configurations
(4) Characteristic of STL storable propellant configurations

[redacted]

The considerations of abort during lunar deboost do not impose additional velocity requirements on the spacecraft. The minimum altitude reached during abort is 600 feet for a shutdown of the deboost stage occurring at an altitude of 1750 feet. The velocity increment required from the return stage to recover to a 50,000 feet altitude orbit is always less than the takeoff velocity increment so that considerable propellant margin exists if an abort occurs.

3.3.1 Velocity Margins

In addition to the nominal velocity increments specified by NASA, a margin of 10 percent velocity capability is required to cover dispersions in vehicle and propulsion characteristics and to provide mission flexibility by allowing a wider choice of lunar landing sites and larger launch windows. The margin required to cover the vehicle and propulsion dispersions has been computed to be no greater than 2 percent of the nominal velocity. A summary of the spacecraft weight and propulsion system dispersions used in the computation is shown on Table 3-3 together with the resulting velocity dispersions.

The 10 percent margin has been used by STL to size propellant tanks and propellant has been loaded to both the 2 percent and 10 percent margins in the performance computations. It is believed, however, that the 10 percent velocity margin required by NASA is arbitrary and should not be considered independently of the other requirements on the system. The very large payload penalty that results from its use suggests that the mission flexibility it affords should be subjected to careful scrutiny.

3.4 SPACECRAFT PAYLOAD PERFORMANCE CAPABILITY

Table 3-4 presents the payload performance capability of five spacecraft configurations using various combinations of pump and pressure-fed cryogenic and storable propellants. The payload weights are shown for an injected weight of 90,000 pounds and for the 2 and 10 percent velocity margins discussed in Section 3.3. The requirement of a 10 percent margin is seen to produce a payload penalty between 1600 and 2000 pounds. The tank sizing criteria used in the computations permit propellant loading to the nominal velocity increments plus the 10 percent margin.

Table 3-3. Spacecraft Performance Uncertainties - 3σ

Parameter	H_2-O_2 Pump Fed Deboost	H_2-O_2 Pump Fed Landing and Return	Storable Pump Fed Landing and Return
A. Parameter Uncertainties			
Specific impulse (sec) (for engine variations)	2.887	5	4
Specific impulse (sec) (for mixture ratio shift)	3.25	3.25	0.40*
PU system residuals (lb)	241	85	168
Inert weight (lb)	214	165	121
Available propellant (lb)	223	92	114
B. Velocity Dispersions			
Specific impulse (for engine variations)	62.4	97.7	112.1
Specific impulse (for mixture ratio shift)	70.3	63.5	11.2
PU system residuals	71.7	66.1	129.8
Inert weight	31.8	63.3	58.3
Available propellant	32.7	36.6	32.0
Guidance	75.5	---	---
Total velocity margins (RSS)	145.7	152.6	184.3
% Nominal velocity increment	1.64	1.66	2.01

* Equivalent to 2% mixture ratio uncertainty.

Table 3-4. Spacecraft Payload Capability

	Configuration	Loading Criteria	Available Payload Weight	Payload Penalty
(1)	H_2-O_2 Pump Fed Deboost	ΔV + 2%	12, 452	2017
	H_2-O_2 Pump Fed Land and Return	ΔV + 2%		
(6)	H_2-O_2 Pump Fed Deboost	ΔV + 10%	10, 458	
	H_2-O_2 Pump Fed Land and Return	ΔV + 10%		
(2)	H_2-O_2 Pressure Fed Deboost	ΔV + 2%	10, 871	1918
	H_2-O_2 Pressure Fed Land and Return	ΔV + 2%		
(2a)	H_2-O_2 Pressure Fed Deboost	ΔV + 10%	8, 953	
	H_2-O_2 Pressure Fed Land and Return	ΔV + 10%		
(3)	H_2-O_2 Pump Fed Deboost	ΔV + 2%	9, 570	1687
	Storable Pump Fed Land and Return	ΔV + 2%		
(3a)	H_2-O_2 Pump Fed Deboost	ΔV + 10%	7, 883	
	Storable Pump Fed Land and Return	ΔV + 10%		
(4)	H_2-O_2 Pump Fed Deboost	ΔV + 2%	8, 261	1687
	Storable Pressure Fed Land and Return	ΔV + 2%		
(4a)	H_2-O_2 Pump Fed Deboost	ΔV + 10%	6, 574	
	Storable Pressure Fed Land and Return	ΔV + 10%		
(5)	H_2-O_2 Pressure Fed Deboost	ΔV + 2%	7, 665	1603
	Storable Pressure Fed Land and Return	ΔV + 2%		
(5a)	H_2-O_2 Pressure Fed Deboost	ΔV + 10%	6, 052	
	Storable Pressure Fed Land and Return	ΔV + 10%		

The lunar spacecraft injected by the launch vehicle includes the propulsion stages and the payload which consists of the command module and its associated support equipment. This latter equipment is physically located in the return propulsion stage (or service module in the case of Apollo). The weight of the payload is greater at liftoff than upon return primarily because of the expenditure of reaction control propellant for coast phase attitude control and vernier midcourse velocity corrections during the outgoing and incoming legs of the mission. This propellant is regarded as part of the initial weight of the associated support equipment (part of the payload) but its expenditure is accounted for in the spacecraft performance computations. The payload weight capability which is quoted below for the several lunar spacecraft propulsion options is, therefore, the weight of the command module and its associated support equipment at a point just prior to jettison of the return propulsion stage (or service module) and re-entry of the capsule.

The relationship of Saturn injection capability and payload weight is shown on Figure 3-3 for the five configurations. These curves are based on the required 2 percent propellant margin. The figure also shows for comparison the required payload weight of a series of command module and associated support equipment payloads. These are discussed in some detail in Section 3. 5.

The structural weights of the spacecraft whose performance is shown on the figure have been adjusted to reflect the variation of injected and payload weight. The landing and return stage structure was, however, sized to accommodate a 154-inch diameter command module. At the lower payload weights this is unrealistic since a smaller command module would be required. Cryogenic and storable propellant landing and return stages were, therefore, also designed to accept a payload of 123 inches in diameter to establish the payload penalty associated with the larger diameter. The effect of this change on the required Saturn injection weight capability is shown on Table 3-5 for a spacecraft designed for a payload of 8200 pounds. This weight was an early estimation of the weight of a 123-inch, 2-man, 8-day mission system.

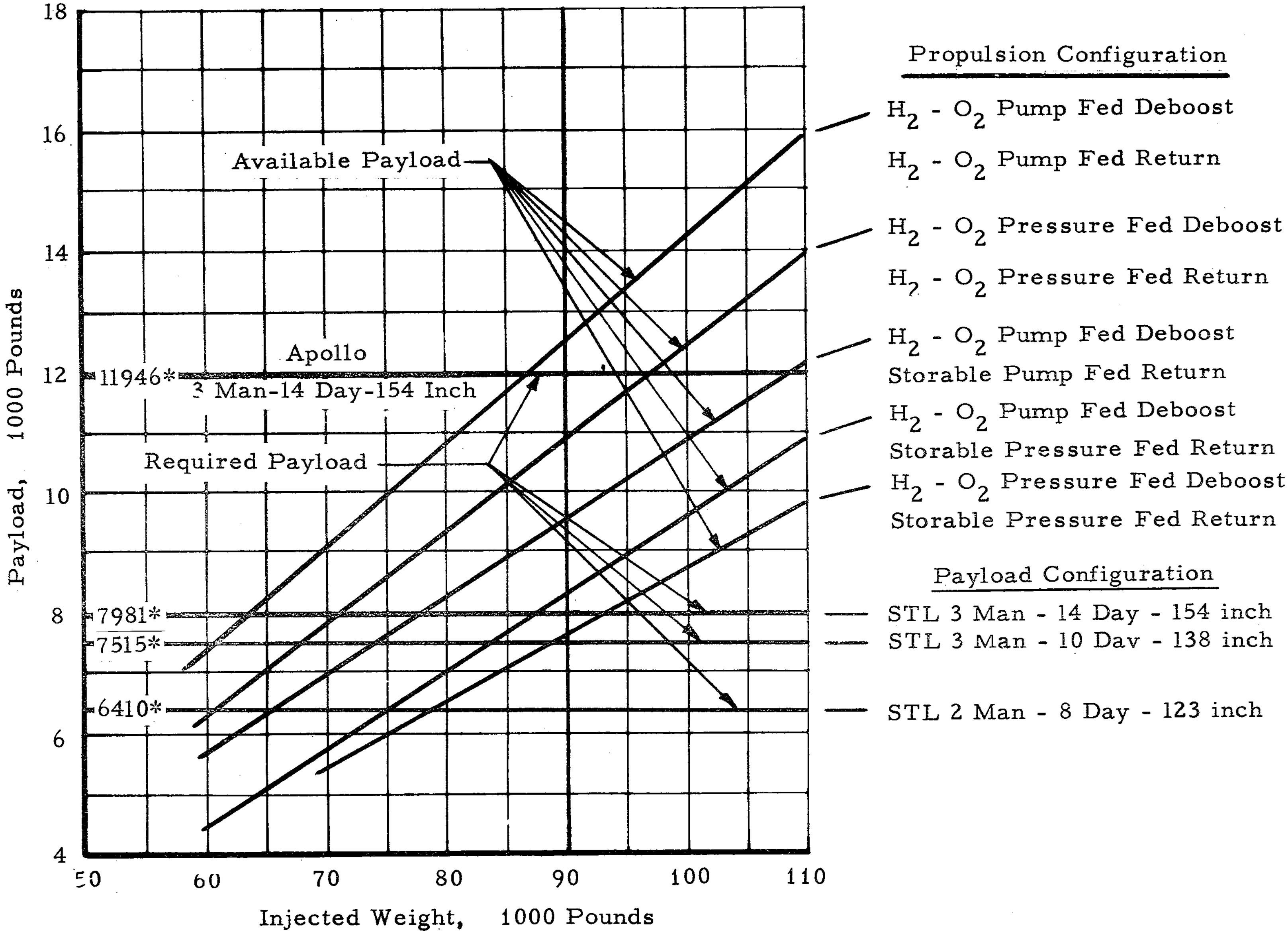


Figure 3-3. Lunar Spacecraft Payload Performance (NASA Velocity Increments + 2% Velocity Margin)

Table 3-5. Effect of Command Module Size - 8200-pound Payload

Propulsion Configuration	Propellant Loading Criteria	Injected Weight Requirement	
		154-inch Capsule	123-inch Capsule
H_2-O_2 Pump Fed Deboost H_2-O_2 Landing and Return	ΔV + 10%	*	74,000
H_2-O_2 Pump Fed Deboost H_2-O_2 Landing and Return	ΔV + 2%	65,000	63,000
H_2-O_2 Pump Fed Deboost Storable Pump Fed Landing and Return	ΔV + 10%	*	85,000
H_2-O_2 Pump Fed Deboost Storable Pump Fed Landing and Return	ΔV + 2%	79,000	78,000

*Not computed but probably about 1000 pounds greater than for the 123-inch case.

3.5 COMMAND MODULE AND ASSOCIATED SUPPORT EQUIPMENT

The required payload of the spacecraft is represented by the command module and the portion of its associated support equipment that is located in other stages. In the STL configurations the bulk of the associated support equipment is located in the upper portion of the lunar takeoff stage. This equipment has been referred to in the past and in this report as the "equipment module" although the equipment components are stowed generally throughout the stage.

Some of the guidance and navigation subsystem equipment which is directly related to lunar terminal guidance and landing is located in the main deboost and landing stages. In addition, 250 pounds of scientific equipment and a stowable airlock (for lunar surface operations) are carried in the landing stage and left on the moon. Table 3-6 summarizes the weight of the 2-man 8-day mission command module and associated support equipment at C-5 liftoff and compares it to the weights of the earlier and revised STL 3-man systems, the NAA Apollo, and to an STL configuration which in size and mission duration is comparable to the NAA Apollo.

For the purpose of determining the weight margin between spacecraft payload capability and command module and associated support equipment weight requirements, the weights shown on Table 3-6 must be modified to reflect weight changes which take place during the missions. The spacecraft payload performance presented in Section 3-4 reflects the payload weight returned to earth rather than the weight at C-5 liftoff. A summary of these payload weights for the STL and NAA systems are presented on Table 3-7. The payload weights are estimated at a point in the mission just prior to jettisoning the lunar takeoff stage before re-entry. The reaction control system propellant expenditures were computed assuming that the midcourse velocity corrections characteristic of the MIT system had been required.

A weight breakdown into major subsystems is shown on Table 3-8 for the 2- and 3-man systems that were studied. The bookkeeping systems used in the present study differs somewhat from that used in the study described in Reference 1. The bookkeeping for the two STL 138 inch diameter configurations are on the same basis as in Reference 1. The STL 123 inch and 154 inch configurations are on the new basis used in the present study. Direct comparisons between individual subsystems cannot, therefore, be made.

Table 3-6. Summary Weight Comparison at C-5 Liftoff

	STL 2 man 8 day 123" diam.	STL (Previous) 3 man 10 day 138" diam.	STL (Revised) 3 man 10 day 138" diam.	NAA Apollo* 3 man 14 day 154" diam.	STL 3 man 14 day 154" diam.
Command Module	4827	6046	5923	8670	6263
Associated Support Equip.					
Lunar takeoff stage	2050	2354	2078	3975	2447
Lunar landing stage	384	276	384	**	384
Main deboost stage	90	88	90	**	90
Subtotal (at C-5 liftoff)	7351	8764	8475	12, 645	9184
Contingency (See Table 3-7)	321	386	375	---	399
Total (at C-5 liftoff)	7672	9150	8850	12, 645	9583

* Based on NAA Weight Statement dated 10 July 1962

** Data not supplied

Table 3-8. Subsystem Weight Comparison

Subsystem	STL 2-man 8-day 123 inch (lb)		STL (Ref 1) 3-man 10-day 138 inch (lb)		STL (revised) 3-man 10-day 138 inch (lb)		NAA 3-man 14-day 154 inch (lb)		STL 3-man 14-day 154 inch (lb)	
	CM	LTOS[2]	CM	LTOS[2]	CM	LTOS[2]	CM	SM	CM	LTOS[2]
Structure and Heat Protection	1834	0	2744	0	2270	0	3850	0	2770	0
Earth Landing	370	0	403	0	433	0	631	0	443	0
Communications and Instrumentation	134	103	91	101	137	103	842	212[6]	137	103
Guidance and Navigation[3]	306	10	263	0	306	10	310	4	306	10
Stabilization and Control	191	0	410	972	191	0	220	0	191	0
Reaction Control	372	678	0	0[8]	362	678	405	1179	362	678
Crew and Crew Support	783	90	988	0	1042	0	1499	0	1137	92
Environmental Control	121	196	379	410	365	413	417	764	177	196
Electrical Power	374	973	334	871	371	874	496	1816	374	1368
Control Panels and Displays	320	0	424	0	424	0	0	0[6]	344	0
Scientific Equipment	22	0[4, 5]	10	0	22	0	0	0[7]	22	0
Totals	4827	2050	6046	2354	5923	2078	8670	3975	6263	2447

(1) All weights are computed at C-5 liftoff.

(2) Associated support equipment in the Lunar takeoff and landing stage.

(3) An additional 207 pounds of guidance and navigation equipment are located in the lunar landing and main deboost stages of the STL configurations.

(4) 250 pounds of scientific equipment are stored in the lunar landing stage and left on the moon.

(5) 22 pounds of scientific equipment are carried to the moon in the command module and 80 pounds are returned in the STL configurations.

(6) The NAA control panels and displays are mostly included in the communications and instrumentation system.

(7) NAA does not show such a category in their 10 July 1962 weight list.

(8) The reaction control system was combined with and listed under stabilization and control.

Table 3-7. Summary Payload Weight Comparison

	STL 2 man 8 day 123" diam.	STL (Previous) 3 man 10 day 138" diam.	STL (Revised) 3 man 10 day 138" diam.	NAA Apollo* 3 man 14 day 154" diam.	STL 3 man 14 day 154" diam.
Command Module	4831	6035	5945	8571	6264
Associated support equipment (LTOS only)	1579	1679	1570	3375	1717
Subtotal	6410	7714	7515	11,946	7981
Contingency (+5%)	321	386	375	---	399
Total required payload	6731	8100	7890	11,946	8380

* Based on NAA Weight Statement dated 10 July 1962

In general, STL applies a 5 percent weight contingency to their payload weight estimates. However, for this study, the command module and associated support equipment weights have individually been derived and identified without the 5 percent contingency included. This has been done since NASA applies its own margins. However, the weight results presented, although achievable, do not represent a conservative estimate. It is STL's opinion that a 5 percent margin is adequate and that a 10 percent margin is conservative. Consequently, a 5 percent contingency margin has been computed, see Table 3-7, and included in the summary weight comparisons for the payload at C-5 liftoff and just prior to earth re-entry.

An attempt has been made to present the NAA Apollo weight breakdown in a manner consistent with STL's, but this may not have been successful in every case.

Table 3-9 summarizes the changes in the weight of the 2-man, 123-inch, 8-day system between liftoff and re-entry. The net change in the command module portion is negligible; the reduction in the associated support equipment weight is 471 pounds. The influence of the type of guidance system used is also shown by comparing the requirements of DSIF and the autonomous MIT systems.

3.6 PAYLOAD MARGINS

The weight increment between available and required payload as shown on Figure 3-3 can be regarded as a margin to take care of system weight growth or booster performance deterioration or to provide mission flexibility through increased velocity capability. The margins shown on Figure 3-3 are presented on Tables 3-10 and 3-11 for the several payload and spacecraft propulsion configurations and a booster injection capability of 90,000 pounds. The margins, presented in pounds and in percent of nominal required payload weight, are over and about the 2 and 10 percent velocity margins discussed previously.

The margins shown in the tables are somewhat higher than presented at the final briefing on this study. This results from the redefinition of required payload discussed in Section 3-5. The differences are most pronounced for the less energetic spacecraft which use storable propellant for lunar landing and return.

[redacted]

Table 3-9. Command Module and Associated Support Equipment Weight Changes Between Liftoff and Re-entry

	Command Module	Associated Support Equipment[1] (lb)[2]	
		DSIF[2]	MIT Guidance[3]
Translunar			
Reaction Control System			
Attitude Stabilization (72 hours)	--	-17.2	-17.2
Reorientations	--	-43.1	-250.8
Vernier Midcourse (10 ft/sec)	--	-93	-93
Lunar Landing			
Reaction Control System			
Attitude Stabilization	--	-18.5	-18.5
Lunar Surface Operations			
Crew and Crew Support System	-54	-20[4]	-20[4]
Scientific Equipment	+58	--	--
Lunar Takeoff			
Reaction Control System			
Attitude Stabilization	--	-4.8	-4.8
Transearth			
Reaction Control System			
Attitude Stabilization (92 hours)	--	-0.4	-0.4
Reorientations	--	-2	-15.8
Vernier Midcourse (30 ft/sec)	--	-50	-50
NET TOTAL - (LB)	+4[5]	-249	-471[5]

NOTES

(1) In lunar takeoff stage
(2) Weight change based on use of the DSIF guidance system configuration
(3) Weight changes based on use of the autonomous MIT guidance system configuration
(4) An approximate number for jettisoned water
(5) Used in the performance computations

[redacted]

Table 3-10. Payload Margins with 2 Percent Velocity Margin
(NASA Velocity Increments - 90,000 pound Injected Weight)

	Configuration									
Deboost	H_2-O_2 Pump Fed		H_2-O_2 Press. Fed		H_2-O_2 Pump Fed		H_2-O_2 Pump Fed		H_2-O_2 Press. Fed	
Return	H_2-O_2 Pump Fed		H_2-O_2 Press. Fed		Storable Pump Fed		Storable Press. Fed		Storable Press. Fed	
	(lb)	(%)	(lb)	(%)	(lb)	(%)	(lb)	(%)	(lb)	(%)
NAA Apollo										
3 man, 14 days	500	4.2	---	---	---	---	---	---	---	---
STL - 154" diam.										
3 man, 14 days	4471	46.0	2890	36.2	1589	19.9	280	3.5	---	---
STL - 138" diam.										
3 man, 10 days	4937	65.6	3356	44.7	2055	27.4	746	9.9	150	2.0
STL - 123" diam.										
2 man, 8 days	6042	94.2	4461	69.7	3160	49.2	1851	28.9	1255	19.6

Table 3-11. Payload Margins with 10 Percent Velocity Margin
(NASA Velocity Increments - 90,000 pound Injected Weight)

	Configuration									
Deboost	H_2-O_2 Pump Fed		H_2-O_2 Press. Fed		H_2-O_2 Pump Fed		H_2-O_2 Pump Fed		H_2-O_2 Press. Fed	
Return	H_2-O_2 Pump Fed		H_2-O_2 Press. Fed		Storable Pump Fed		Storable Press. Fed		Storable Press. Fed	
	(lb)	(%)	(lb)	(%)	(lb)	(%)	(lb)	(%)	(lb)	(%)
NAA Apollo										
3 man, 14 days	---	---	---	---	---	---	---	---	---	---
STL - 154" diam.										
3 man, 14 days	2577	32.3	972	12.2	---	---	---	---	---	---
STL - 138" diam.										
3 man, 10 days	2943	39.2	1438	19.1	368	4.9	---	---	---	---
STL - 123" diam.										
2 man, 8 days	4048	63.0	2543	39.6	1473	23.0	164	2.5	---	---

3.7 SPACECRAFT PROPULSION STAGE DESIGN

The lunar spacecraft configuration studies were largely concentrated on vehicles using pump fed cryogenic propellants. The general arrangements are, however, directly applicable to pressure fed systems and to storable propellants. The propellant tank sizes and stage lengths would change but the general arrangements would be preserved.

The configurational arrangements discussed in this section are quite similar to those used in the earlier study. The additional effort permitted by the present study has resulted in some design refinement and in a more detailed solution of design problems.

3.7.1 Launch Configuration

The Saturn C-5 configuration for the lunar direct mission is shown on Figure 3-4. The basic boost vehicle consists of the S-IC lower stage, an S-II second stage and an S-IVB third stage (260 inch diameter). A short adapter section has been added to the forward end of the S-IVB with a manufacturing splice at station 3236. This adapter extends to station 3270 where the lunar spacecraft is attached with a separation joint. This joint was established as far forward as practical to minimize spacecraft weight after separation. The launch escape system completes the vehicle.

A general arrangement of the spacecraft and launch escape system is presented on Figure 3-5. This spacecraft and launch escape system represent a design consistent with a 90,000 pound injected weight. The spacecraft shown uses cryogenic propellants in both the deboost and return stages. A command module size and weight equivalent to the 154 inch diameter Apollo was used for the configuration since it is most nearly compatible with the available direct flight payload capability. The launch escape system utilizes a multiple nozzle solid propellant rocket motor for abort, a dual nozzle solid propellant rocket motor for separating the tower from the command module and a single nozzle solid propellant "kicker" rocket for

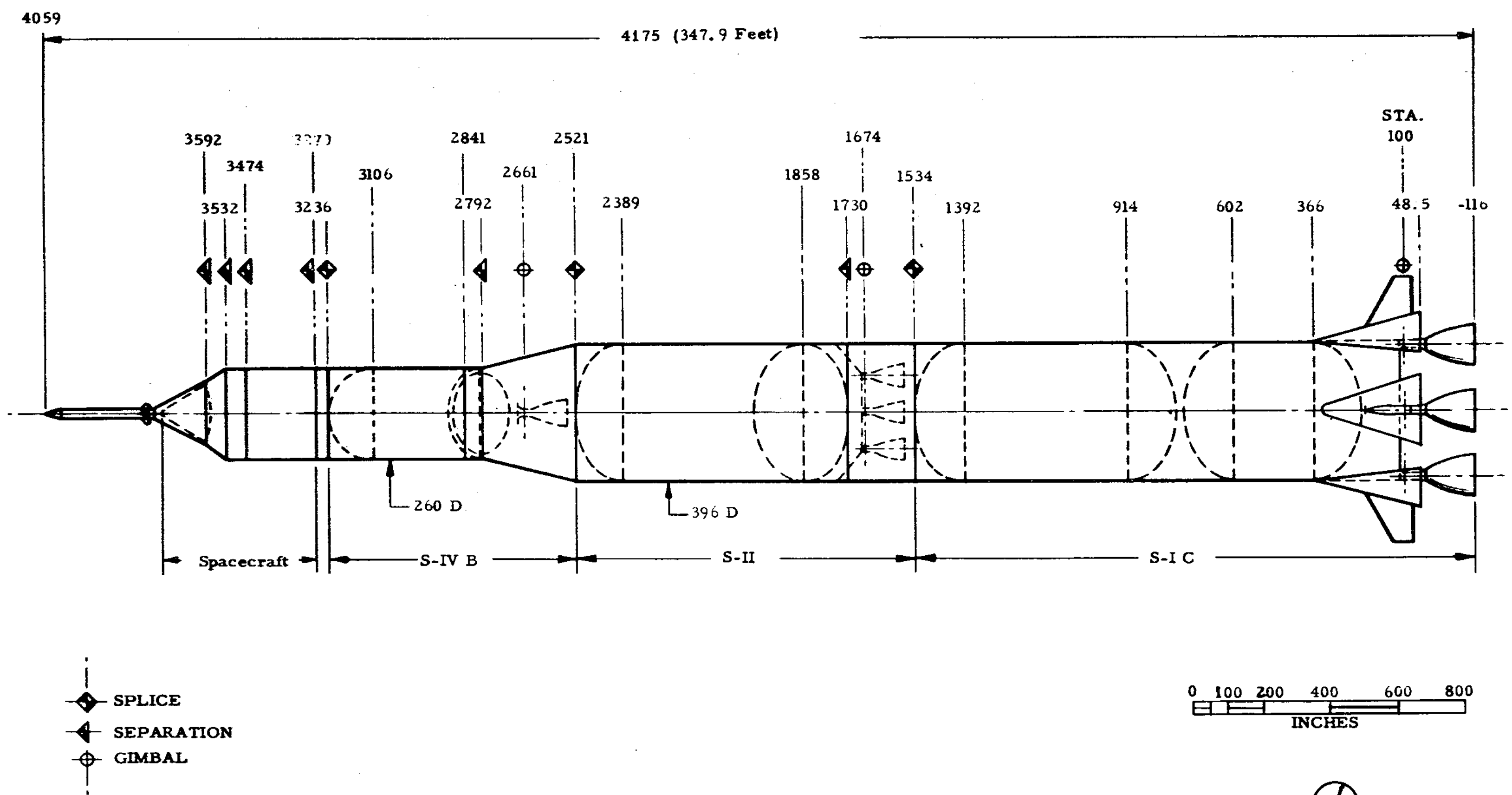


Figure 3-4. Direct Approach Configuration, Saturn C-5

SPACE TECHNOLOGY LABORATORIES, INC.
A SUBSIDIARY OF THOMPSON RAMO WOOLDRIDGE INC
ONE SPACE PARK · REDONDO BEACH, CALIFORNIA

OCT. 3, 1962 PD36-07C

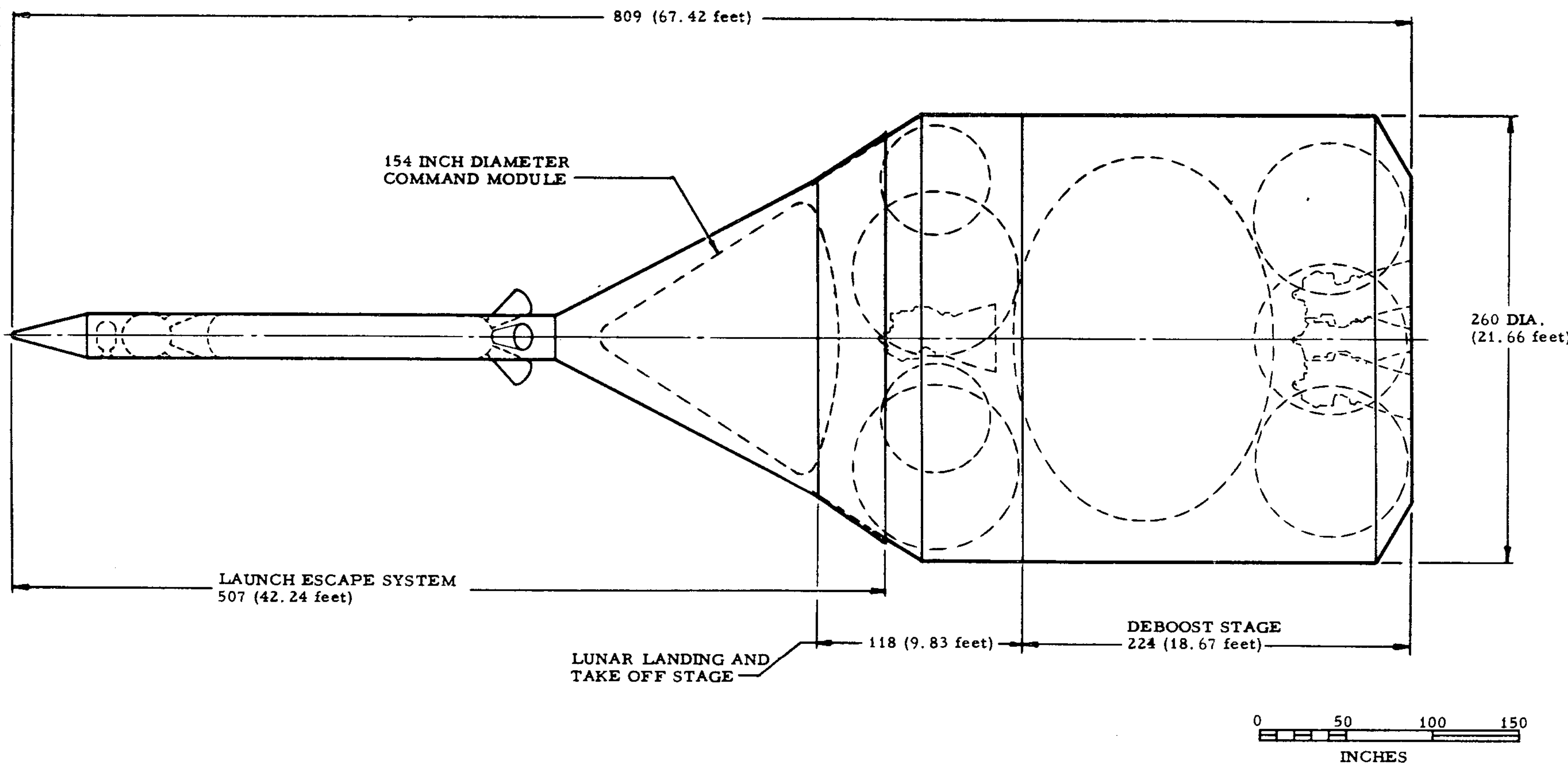


Figure 3-5. General Arrangement, Spacecraft and Launch Abort System (154-Inch Diameter Command Module, Cryogenic Propellants)

SPACE TECHNOLOGY LABORATORIES, INC.
A SUBSIDIARY OF THOMPSON RAMO WOOLDRIDGE INC
ONE SPACE PARK • REDONDO BEACH, CALIFORNIA
OCT. 15, 1962
PD 36-58

lateral separation clearance. NAA Apollo data was utilized in the sizing and design of the abort rockets.

A launch escape system fairing was carried aft from the rocket base and covers the entire command module in a manner similar to that used in the earlier direct flight study (Reference 1). This arrangement differs from the Apollo system which uses an open truss structure which attaches to the command module. Aerodynamic stability investigations have led to the addition of a flared skirt at the aft end of the fairing to provide static stability during abort. The primary advantage obtained by this system is to reduce the required command module structural weight since all bending and shear loads are carried by the fairing, and the command module conical surface is not subjected to high dynamic pressures during the boost or abort phases. Further, the system can be made aerodynamically stable so that tumbling is eliminated. The abort system weight is somewhat heavier but results in an insignificant reduction in allowable payload weight since the tower is jettisoned early in the S-II portion of flight.

3.7.2 Lunar Spacecraft Arrangement

The lunar spacecraft configuration remains unchanged from injection until the deboost stage has brought the vehicle to rest about 1000 feet above the lunar surface. At this time the empty deboost stage is jettisoned. The lunar landing vehicle then performs the final descent, hover, translation and touchdown operations required for the lunar landing. Lunar landing propulsion is provided by the throttlable engine system of the lunar takeoff stage. The actual landing stage consists basically of a detachable structure to which the landing shock attenuation system is attached.

Upon completion of the lunar stay, the lunar takeoff vehicle launches from the landing stage, leaving it on the lunar surface. The lunar takeoff stage completes the injection into the earth return transit. Just prior to earth

[redacted]

re-entry the command module is separated to complete the guided re-entry and recovery phases.

Configurations with a non-jettisonable deboost stage are feasible. However, their payload capability is considerably reduced because of the penalties paid in increased weight of landing gear and landing load carrying structure. This results in part from the heavier weight at landing and in part from the higher center of gravity of these configurations. Extendable landing gear are required to provide a stable landing configuration.

The deboost stage inboard profile for a cryogenic propellant design is shown on Figure 3-6. The basic configuration is quite similar to that used in the previous study (Reference 1). The designs utilize a large ellipsoidal tank for the liquid hydrogen and three spherical tanks for the liquid oxygen. The tanks are insulated to minimize propellant boiloff.

Three P and W gimballed RL10A-3 engines are mounted in a symmetrical pattern about the stage centerline. The engines are operated in the normal full thrust pump fed mode to produce the large velocity increments needed for lunar orbit circularization and for deboost. They are also operated in a low thrust (ullage) mode, using tank pressurization and bypassing the pumps, to produce the smaller increments needed for propellant settling, midcourse velocity corrections, and the Hohman transfer retro increment. This arrangement was chosen over the three canted and fixed engines used in the earlier study to eliminate the complexity of the two gimballed attitude control auxiliary engines and their separate earth storable propellant tankage and system provisions.

The deboost stage shell structure efficiency has been improved and the engine and oxygen tank support beams have been changed from a truss to a web beam structure. The cryogenic tank attachments have also been improved to reduce heat input to the propellants. The double wall outer structural shell and an enclosure at the aft end of the stage provide micrometeoroid protection.

[redacted]

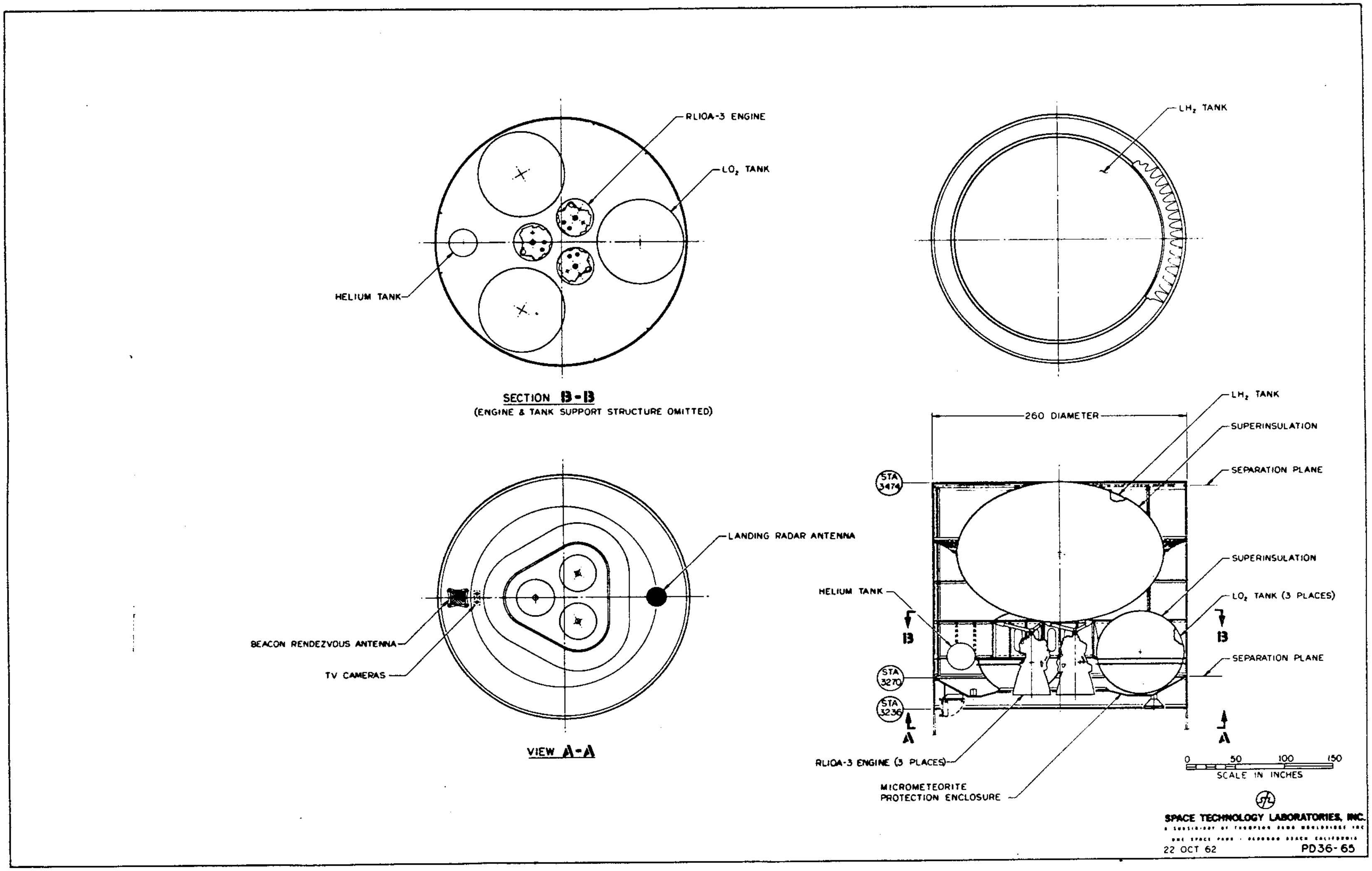


Figure 3-6. Inboard Profile, Deboost Stage (Cryogenic Propellant)

Lunar landing radars and their antennas, as well as television cameras are located at the base of the deboost stage. Considerable emphasis was placed on design simplicity and accessibility. Maximum interchangeability of components and assemblies was provided throughout, e. g. the three engine mount trusses are identical to each other. Similarly, the oxygen tanks (and their mounts), subassemblies for the three central support beams, and outer shell panels were designed for multiple usage, etc. The lower tanks and plumbing may be installed and checked out and the engines aligned prior to installing the ellipsoidal hydrogen tank. The hydrogen tank supports can be installed from the upper end and there is ample room to make the plumbing close out connections from the bottom. Each engine may be removed and replaced individually without affecting the others. The micrometeoroid protection enclosure is installed last with fasteners and may be removed at any time for further access. The entire stage is mated to the Saturn C-5 at station 3236 with all attachment access being from the exterior.

After deboost stage separation at about 1000 feet altitude the vehicle is landed using the single throttlable and movable engine of the takeoff stage propulsion system. The inboard profile drawing for the lunar landing and takeoff stages is shown on Figure 3-7. The tankage is sized for the maximum landed payload capability associated with a 90,000 pound injected weight. Consequently, the stage has been designed to mate with an Apollo 154 inch diameter command module and abort fairing since this vehicle weight most closely approaches the allowable payload capability.

As in the earlier study the diameter of the landing stage has been maintained at 260 inches. This results in a squat landing configuration with a low center of gravity which does not require extendable landing gear for lunar landing stability. Several tankage arrangements including cylinders, spheres, toroids, ellipsoids, etc., were investigated to determine the optimum compromise between structure and tank weight. Spherical tanks for both

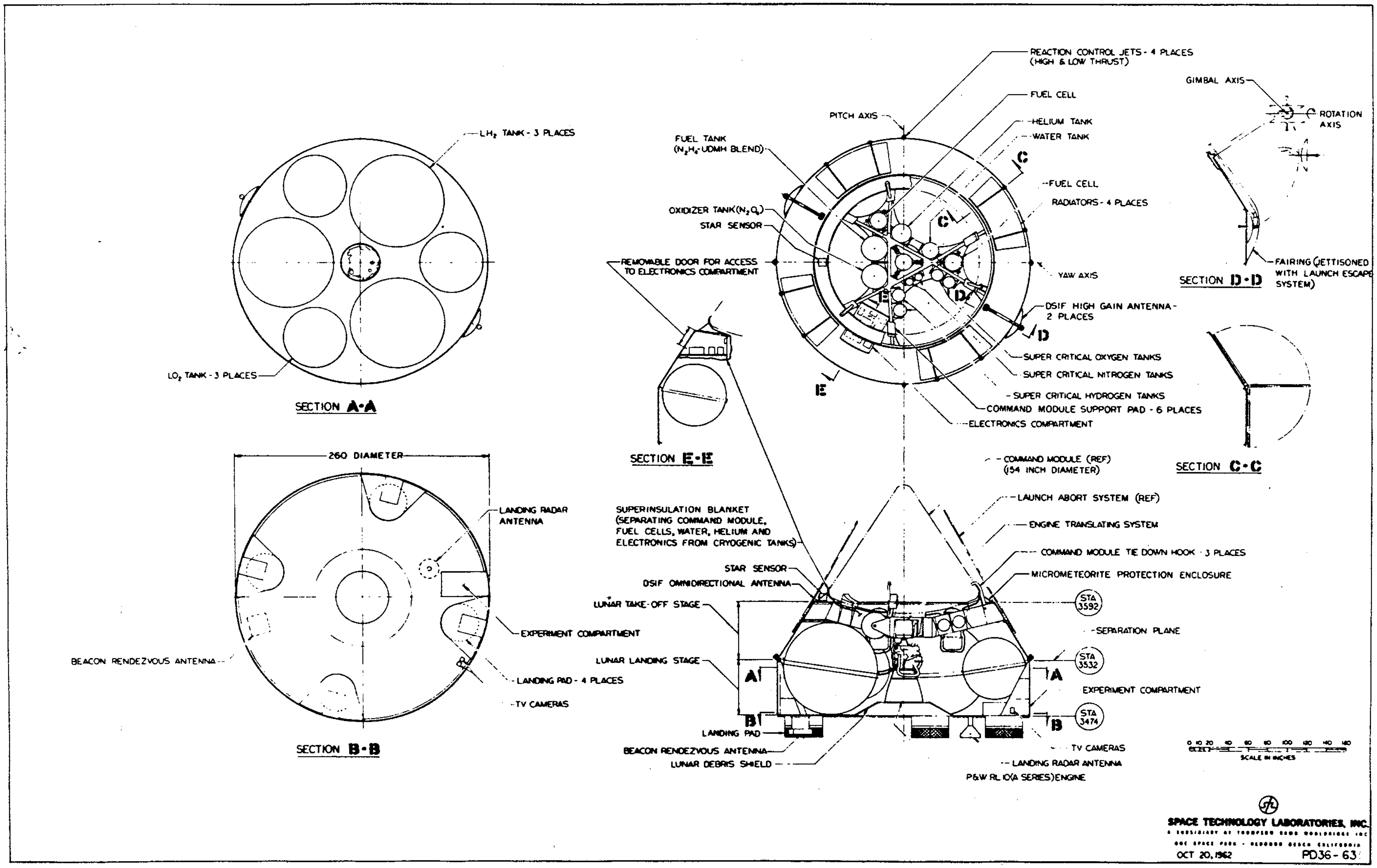


Figure 3-7. Inboard Profile—Lunar Landing and Take-off Stages

the oxygen and hydrogen were found to be the most efficient when the effects on the outer shell and supporting structure and the landing stability considerations were taken into account. The engine support beam, tank support and landing pad support structural efficiency ~~has~~ also been improved over the earlier study. Micrometeoroid protection is incorporated in the double wall shell structure.

Two major configuration changes were made in the design of the previous study. The first was a reduction in the number of lunar landing pads from six to four and a reduction in the pad depth with a consequent weight reduction. This change resulted from the use of the Apollo work statement lunar landing criteria which is less severe than the criteria used by STL in the previous study. The reduction in landing velocities, in combination with a lower center of gravity, made it possible to achieve satisfactory landing stability with four pads. The reduced landing acceleration resulted in decreased pad depth and a weight saving.

The second change was the elimination of the three vernier landing engines used for terminal lunar landing and attitude control. The new design uses the lunar takeoff engine for landing and a combination of engine translation and auxiliary reaction control jets for attitude control. This change reduced the propulsion system complexity and provided a substantial improvement in performance.

The landing and takeoff engine thrust orientation utilizes translation rather than gimballing for control since the center of gravity is near the gimbal point thus making gimballing ineffective. Several well known mechanical principles can be applied to the translator although a detail study was not conducted to determine the most optimum approach. The central beams support the engine, propellant tanks, subsystem equipment items and the command module. Command module tie-down hooks and umbilical connections are located at the three beam terminal points.

An electronic equipment compartment and a scientific equipment storage compartment are accessible from the exterior of the stage. A superinsulation blanket separates the command module, fuel cells, water, helium and electronics from the cryogenic propulsion system tanks to minimize heat transfer to the propellants. This blanket is also useful in maintaining the command module support equipment compartment temperature at a level which is compatible with the command module phenolic nylon heat shield material. Four controllable and deployable environmental control system thermal radiators are located on the exterior of the structural shell. Two erectable and steerable DSIF antennas are similarly stowed on the outer periphery. Both the radiators and the antennas are protected from high dynamic pressures during boost by fairings which are retained at the upper end by the launch escape system fairing skirt and released by the jettisoning of the abort system.

Lunar landing radar antennas are located at the bottom of the stage and are level with the bottom of the landing pads. This location was selected to prevent the landing pads from adversely affecting the antenna patterns. Since the units are not reused after landing, it was considered simpler to allow them to strike the surface than to complicate the antenna and electronics system. Lunar landing television cameras and landing lights are also located in the lower area of the stage. A debris shield covers the bottom of the landing stage to protect the components from lunar surface material that might be thrown up by the engine blast impingement. The need for the debris shield and its effectiveness as well as the required ground clearance distance for the engine cannot be realistically assessed until better lunar surface data are obtained. The design proposed represents a suggested approach which is reasonable according to our present knowledge of the lunar surface.

[redacted]

3.8 COMMAND MODULE AND ASSOCIATED EQUIPMENT DESIGN

3.8.1 Command Module Configuration

Extensive design and parametric studies were carried out to determine the relationships between command module size and the arrangement of crew and equipment. The objective of these studies was to permit the selection of a command module size which would provide adequate usable volume for the crew while minimizing system weight. In general, the weight of command module structure and heat shield increases approximately as the square of its diameter while the usable crew volume increases somewhat faster than the cube of the diameter.

The parametric study also considered the use of large external airlock structures to provide additional crew volume, since on a volumetric basis space structures are more efficient than re-entry vehicles. Figure 3-8 shows the type of configurations considered. Point design studies were made of selected configurations to ensure a consistent and realistic treatment of equipment installation volumetric requirements, structural weight, pressurization system, recovery and earth landing system weights.

During the 3-man lunar direct flight study (Reference 1), STL examined the capsule size selection from the point of view of minimum volumetric requirement. It was assumed that highly trained and motivated personnel would be selected for the task. Further consideration of crew work station functions and the volume required for crew mobility indicated that 30 to 35 cubic feet of usable volume was adequate for seated work stations and that mobility requirements merely required adequate room to stand erect, to stoop, squat and generally mobilize the various joints and muscles. The 3-man 138-inch diameter configuration provided 36 cubic feet for each seated crew station and 60 cubic feet for the center station with the seat stowed. The center station provided considerably more space than required for crew mobility. The crew volume provisions of the 3-man configuration

[redacted]

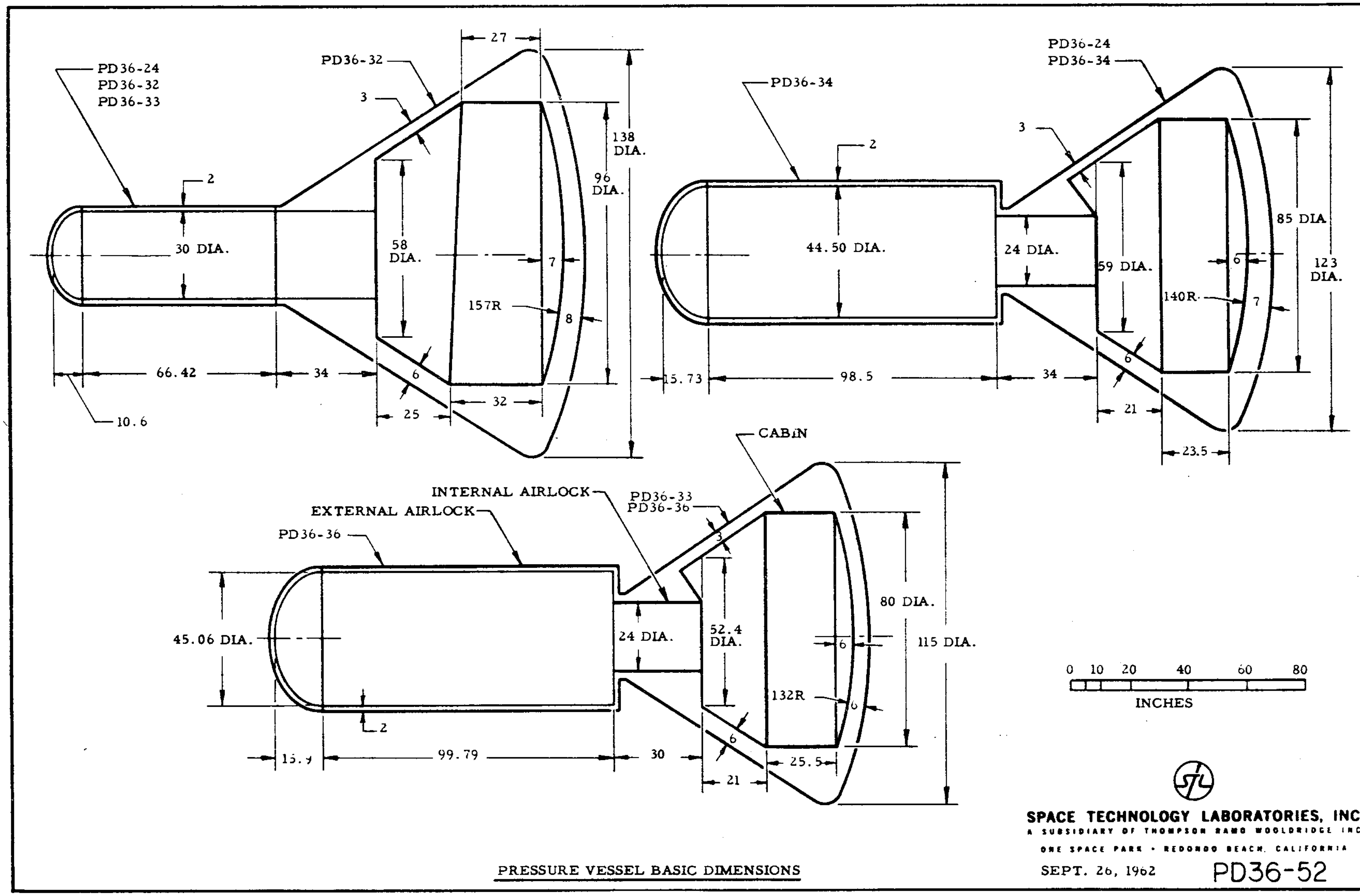


Figure 3-8 Pressure Vessel Basic Dimensions

were considered by STL to be quite adequate, even without the external airlock (Figure 3-9), with an average usable crew volume of 44 cubic feet per man.

Research on the minimum crew space habitability problem conducted by NASA at the Ames Research Center (Reference 6) has provided some interesting new data for consideration. A simulated 2-man capsule containing a total volume of 149.50 cubic feet was used in the experiment. Deducting the fixed equipments left a gross volume available to the crew of 123 cubic feet or an average of 61.5 cubic feet per man. From Figure 1 of Reference 6 STL calculated a usable volume of 48 cubic feet for the seated crew station and 58 cubic feet for the reclining crew station. A minimal standing space was provided at the rear of the seats. Two highly motivated subjects spent a 7-day work schedule in the test capsule. The performance efficiencies of the subjects were good except as affected by the 4-hour duty cycle selection which could be improved by the split duty cycle routine used by the military.

Upon completion of the experiment it was concluded that confinement of two men in a 123 cubic feet volume for 1 week was completely tolerable. The physiological deterioration was of the same nature as that expected from a week of bed confinement, although less extensive. Both subjects felt they could have continued for another week without voluntary performance deterioration. The test pilot subject felt that the capsule volume could be further reduced if adequate space were provided for mobility (stretching, bending, etc.). The other subject believed that the volume could be markedly reduced (especially for a zero g field) without decreasing performance.

Based upon the above experiment, as well as a review of the parametric configurations and the 2-man study of Reference 7, a 123-inch diameter command module was selected as a logical compromise between crew and equipment requirements and minimum weight. A storable airlock/repressurization mode was selected based on system weight consideration. It

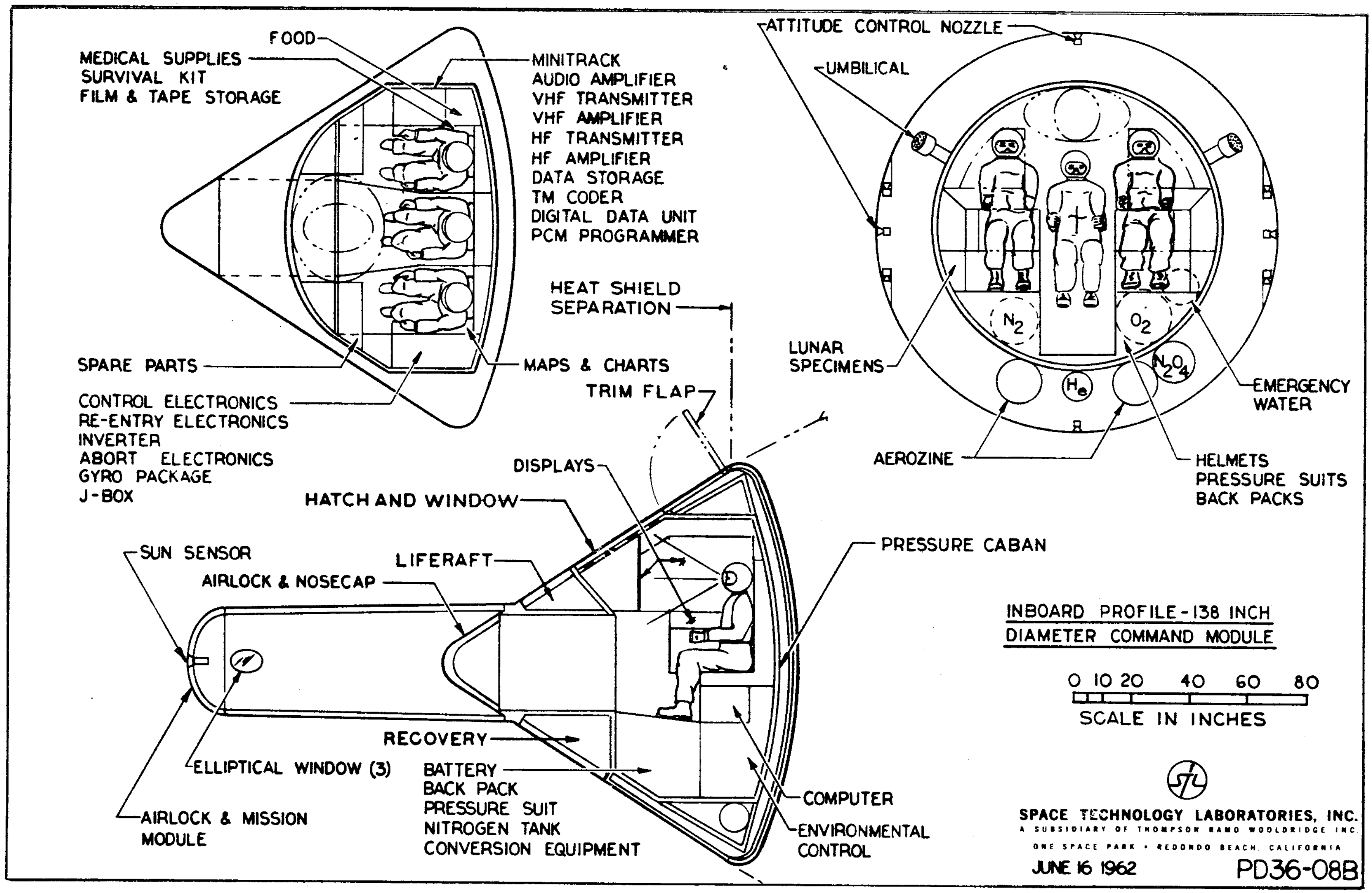


Figure 3-9 Inboard Profile—138-Inch Diameter Command Module

utilizes a collapsible metal airlock which is stowed in the lunar takeoff stage until required upon arrival on the lunar surface. It is installed on the command module by the crew. The examination of the internal air-lock and recovery arrangement of previous configurations indicated that both could be improved. Installation of the recovery parachutes in the central area occupied by the airlock in the earlier 3-man configuration (Figure 3-9) provides a much more efficient recovery arrangement at less weight and also provides an emergency exit after earth landing. However, a normal entrance hatch is provided in the cabin side wall.

The placement of the crew within the command module is shown on Figure 3-10 and the command module interior arrangement on Figure 3-11. The relationship of the crew members to each other and to the command module equipment is readily apparent. Considerable attention was given to the interior arrangement to provide maximum crew capatibility, ease of operation, simplicity and crew mobility. Crew safety was a primary consideration throughout the design study.

In the Apollo design, the pressure suits and helmets are stored in the airlock during shirtsleeve environment conditions. For the previous 3-man design, STL stored them in the left and right floor compartments. After careful consideration of the difficulties involved in obtaining and donning the pressure suit in an emergency and under unfavorable conditions, it is felt that other approaches should be devised. It is recommended that a suit arrangement be developed which allows the suit to be opened and remain attached to the contour seat when not in use, thus providing immediate availability. Helmets should also be stored within an arm's reach.

The command module crew stations are shown in Figure 3-12. Standing height is provided for a person of 5 foot 10 inch height while wearing a typical pressure suit. Both crew members may stand simultaneously and face in any desired direction. Adequate space for mobility is available for exercising. Side console height is convenient for both the seated and standing positions.

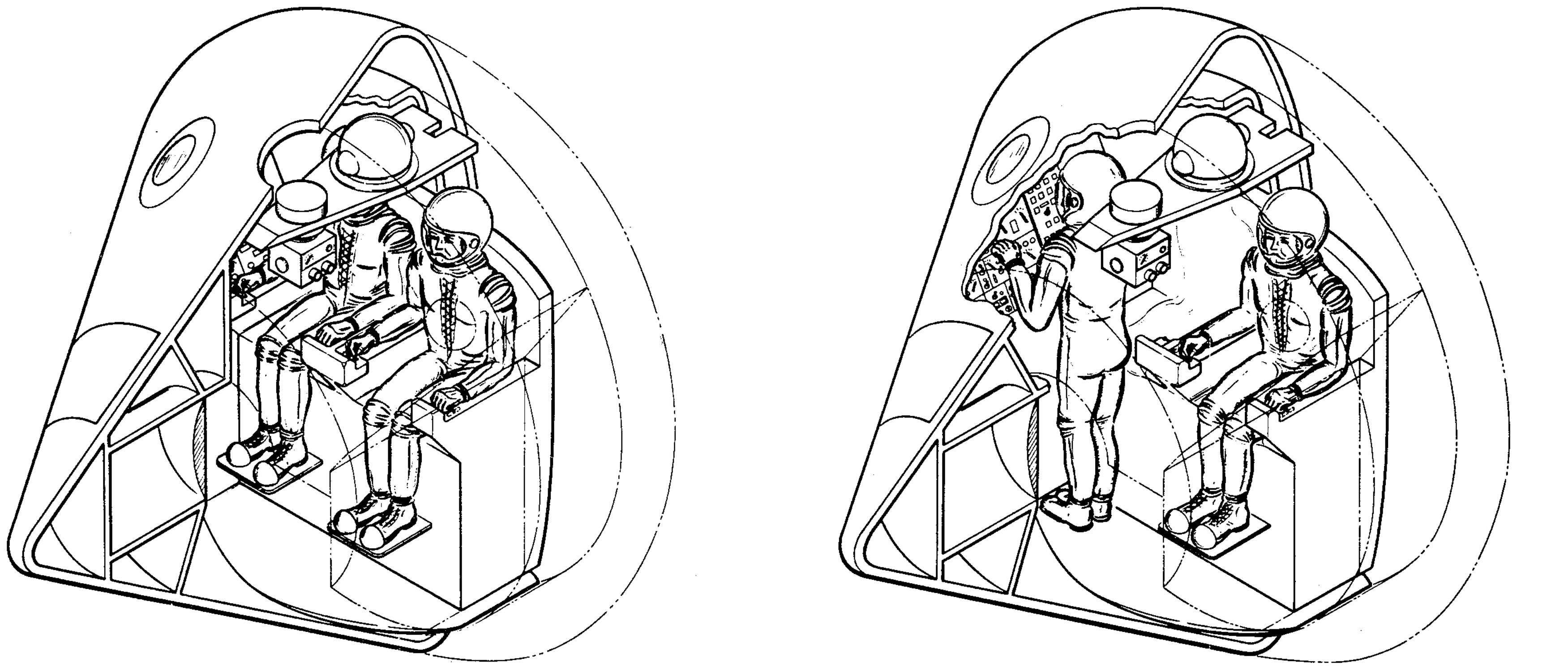


Figure 3-10 Crew Arrangements, Two-Man Command Module

SPACE TECHNOLOGY LABORATORIES, INC.
A SUBSIDIARY OF THOMPSON RAMO WOOLDRIDGE INC.
ONE SPACE PARK · REDONDO BEACH, CALIFORNIA
OCT. 22 1962 PD36-68

[redacted]

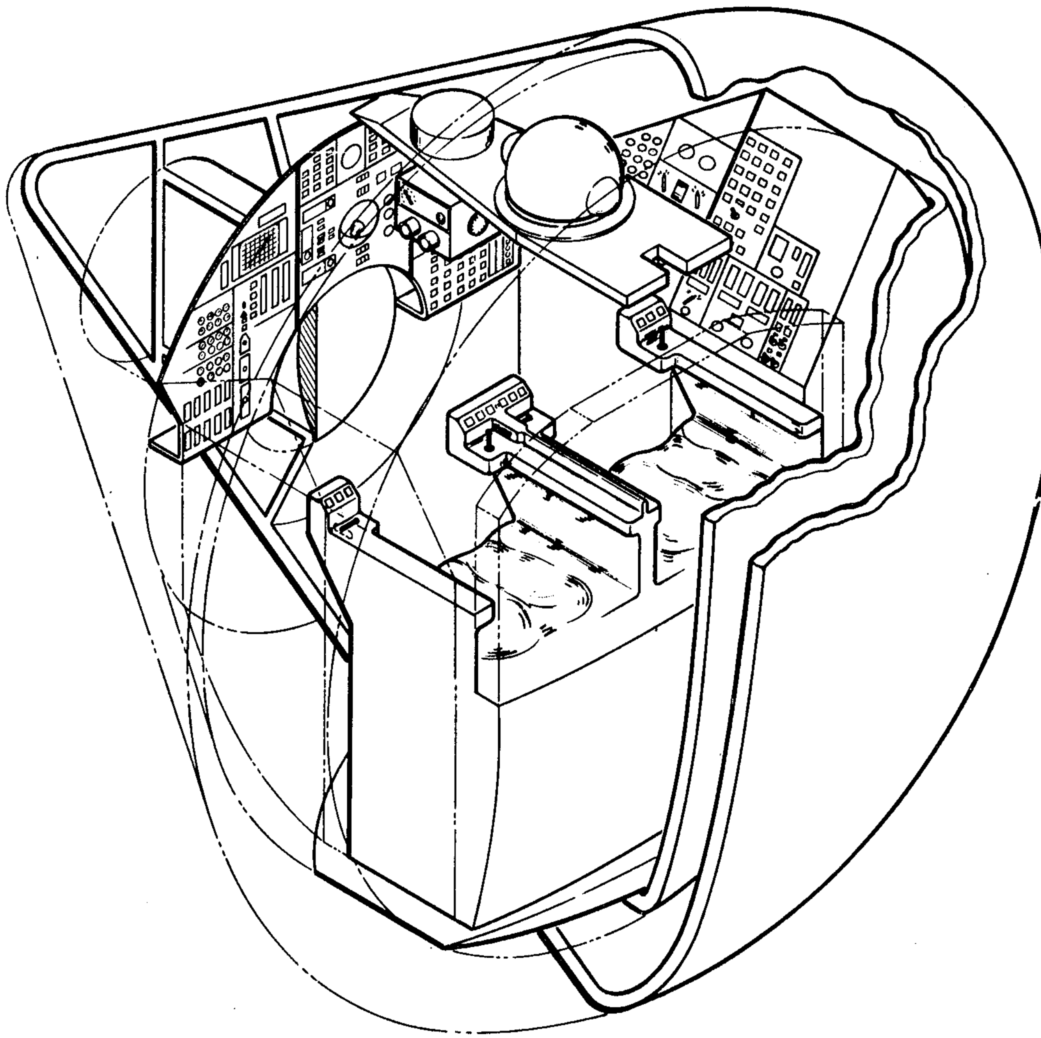


Figure 3-11 Interior Arrangement, Two-Man Command Module

[redacted]

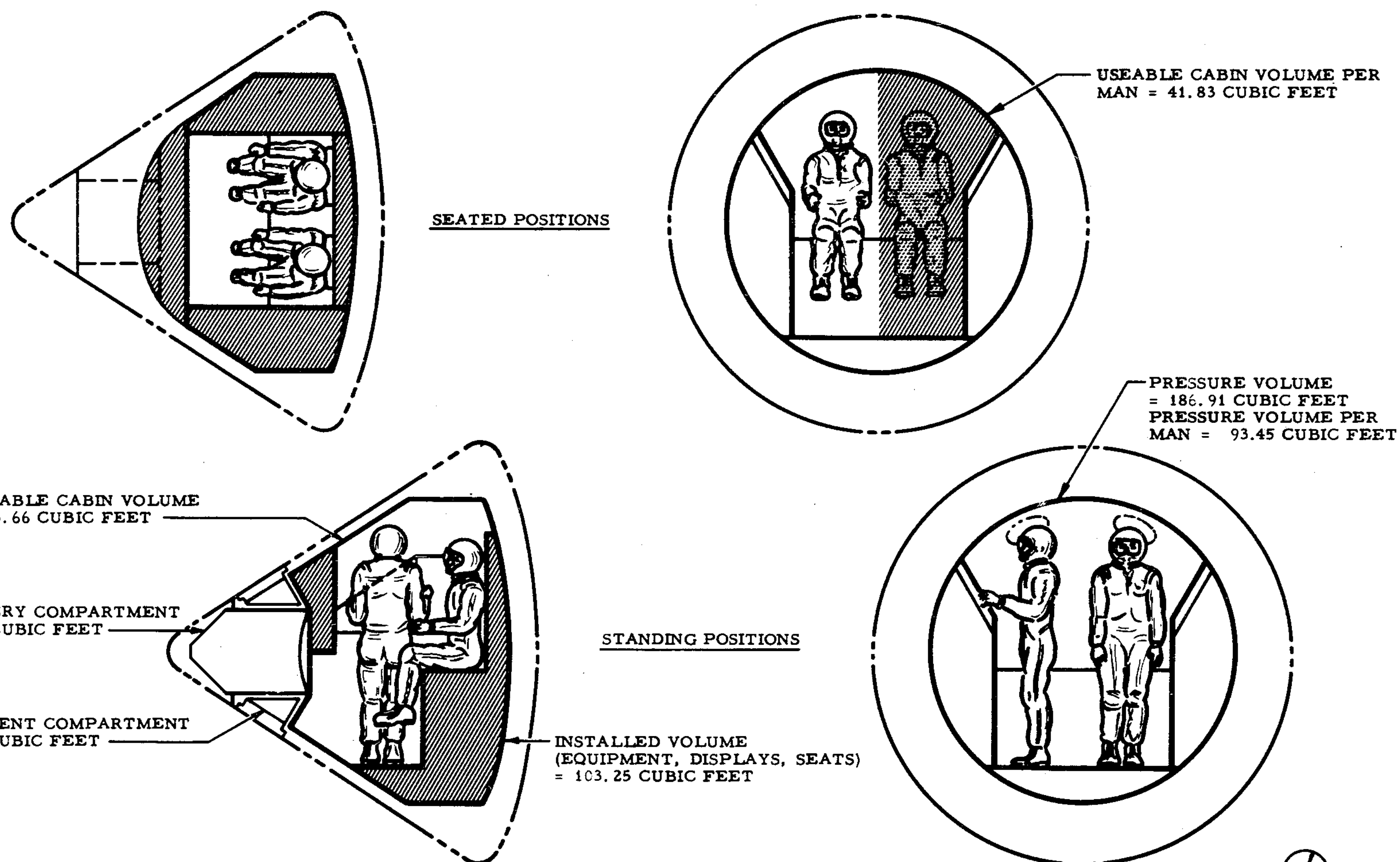


Figure 3-12 Command Module Crew Stations and Volumes (123-Inch Diameter)

A reduced-scale Apollo basic shape was used with the mold line dimensions shown on Figure 3-13. The general configuration is quite similar to the previous STL command module arrangements (Reference 1). However, several refinements in mechanization have been incorporated. The exterior is composed of a phenolic nylon ablative heat shield on the aft end and a reradiant heat shield on the conical portion.

The ablative heat shield is supported by an aluminum honeycomb sandwich structure and is deployable for earth landing. The reradiant heat shield is comprised of a series of Hastelloy stell panels with tapered corrugations running the length of the conical section. These panels are supported on phenolic rings and the edge attachments are designed with sufficient float to prevent restraint during thermal expansion. A teflon ablative sheet is attached with mechanical fasteners to the lower quadrant of the conical section and over the small forward hemispherical dome. This ablative sheet lays outside the basic command module mold line.

The basic arrangement of the crew and command module equipments is shown on the command module inboard profile drawing of Figure 3-14. The crew members are seated side-by-side against the aft pressure cabin bulkhead in non-adjustable seats. Retractable foot rests are used to provide an unobstructed standing area. The crew are restrained by a system designed by STL. Heavy equipment items are located under the seats and the floor to provide an offset center of gravity for earth re-entry. Other equipment items are located under the left and right hand consoles.

Two elliptical shaped windows with major axes of 10 and 12.7 inches are located in the upper quadrant of the conical section. One of the windows is located in the main entrance hatch. Both windows have covers which hinge forward and which can be operated by the crew to reduce the solar heat input during periods where the orientation with the sun is unfavorable. The main access hatch is a 22 by 28 inch ellipse. It opens inward to provide more positive sealing when the cabin is pressurized and easier emergency opening if the command module should sink after a water landing. A remotely operated sun shade is also provided over the periscope.

[redacted]

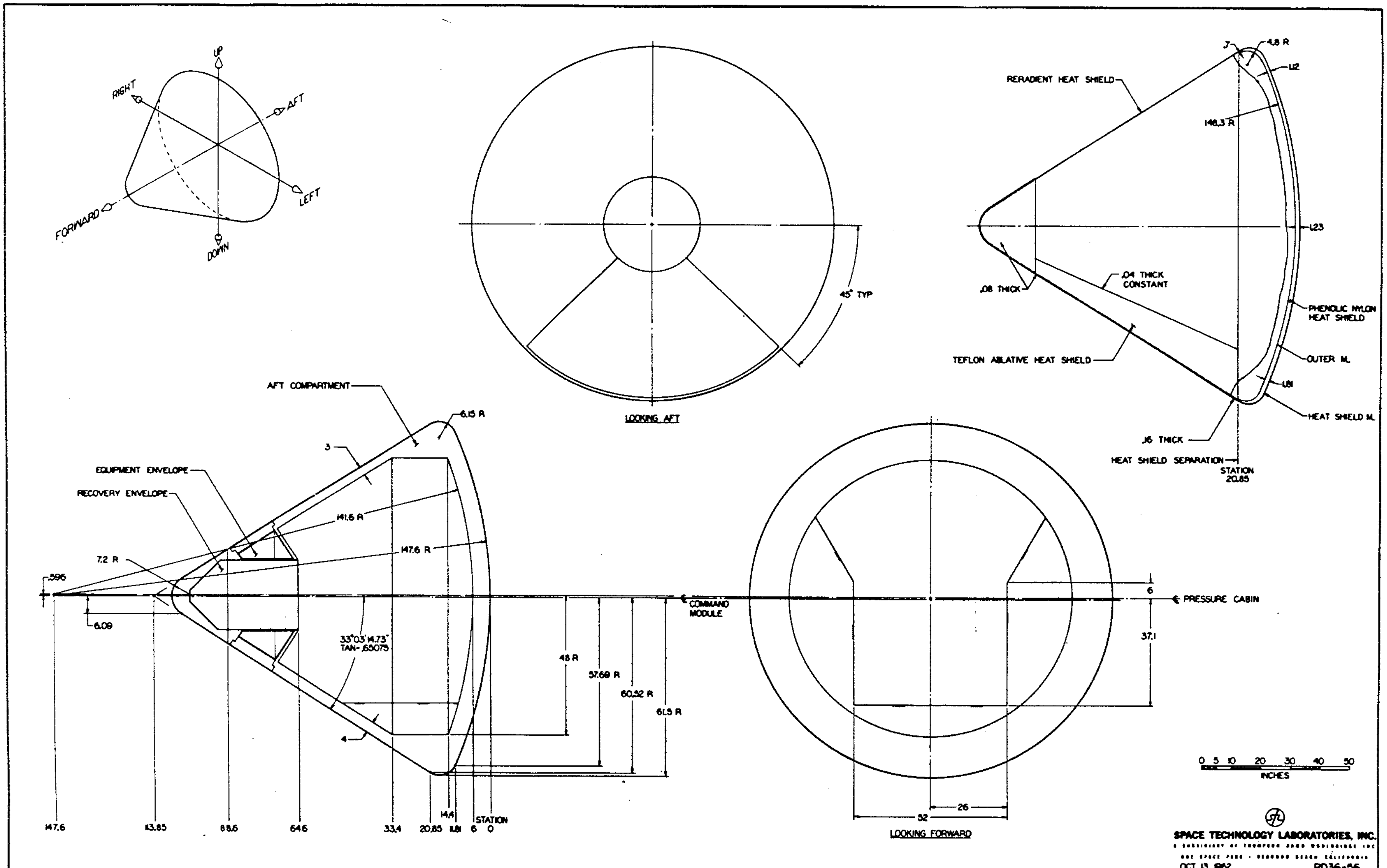


Figure 3-13 Basic Dimensions (123-Inch Diameter Command Module)

[redacted]

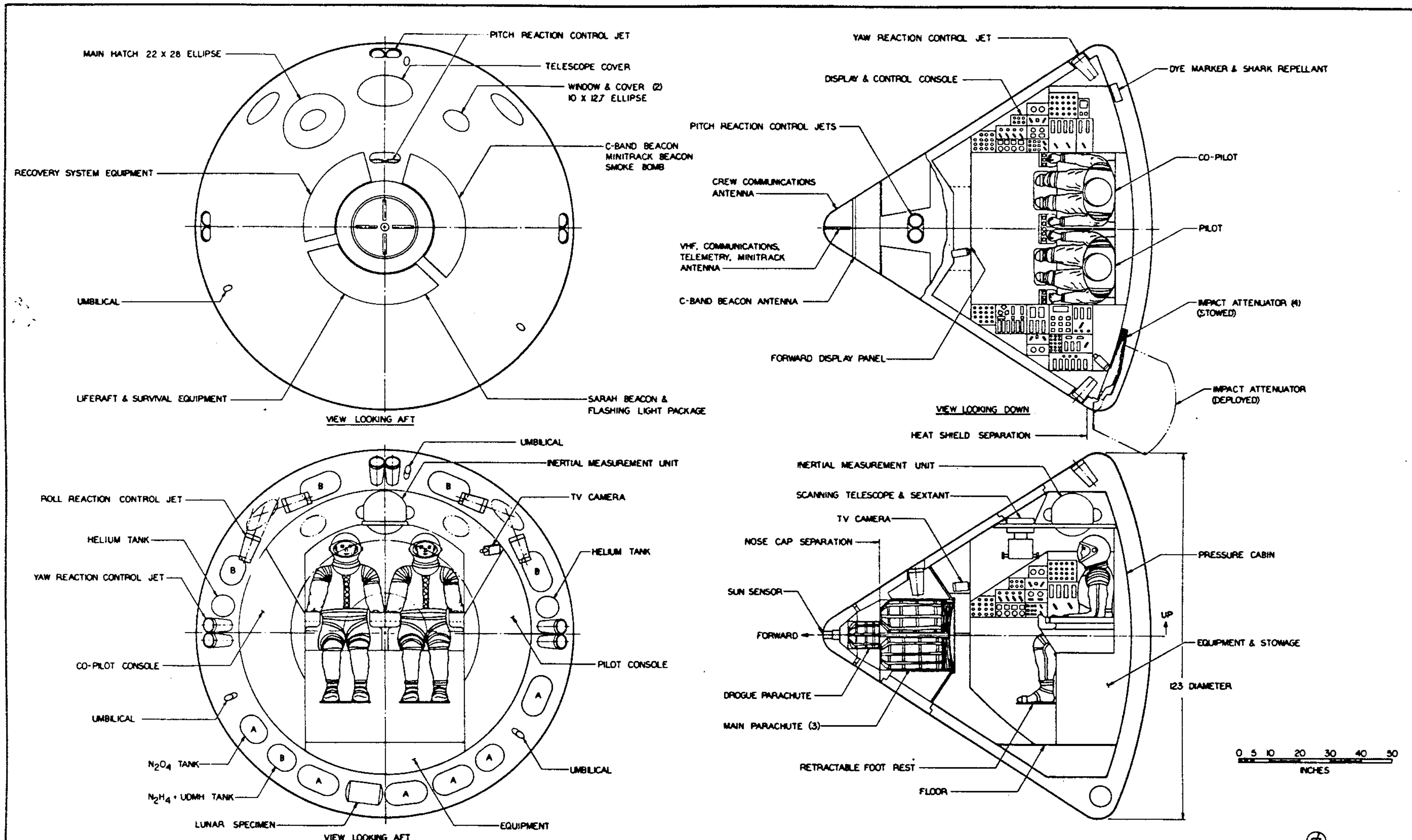


Figure 3-14 Inboard Profile—Command Module

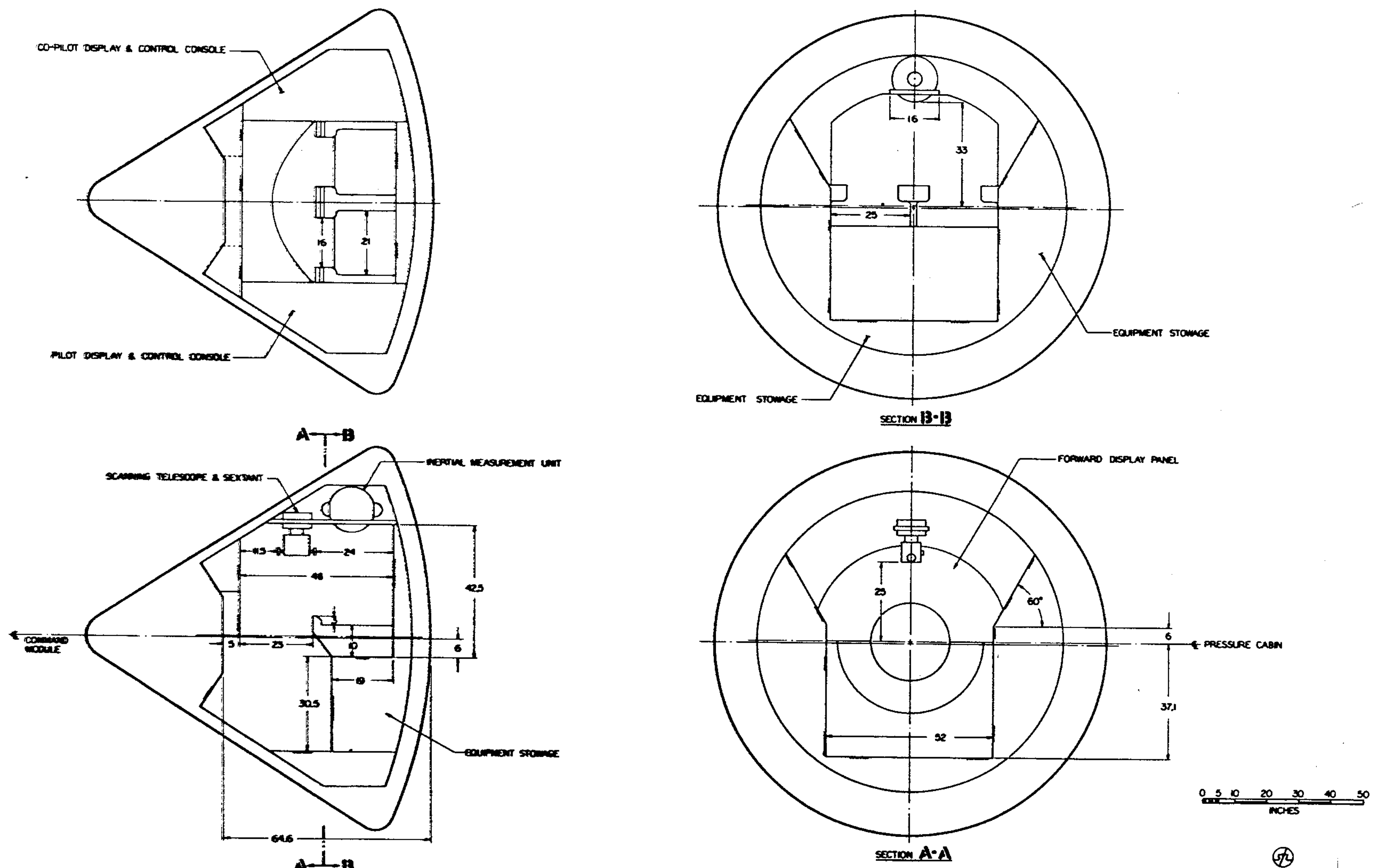
SPACE TECHNOLOGY LABORATORIES, INC.
OCT. 22, 1962
P036-66

The inertial measurement unit and the guidance and navigation periscope are centrally located in an overhead position. This location is considered to be advantageous since the periscope can be used from either crew station and the associated displays on the forward panel are also readily available from either station. The IMU and periscope are mounted on a machined beryllium base for structural alignment stability and may be installed as a unit. Displays are located on a main forward panel and on the left and right consoles. None of the forward displays require hand operations during high "g" conditions. Attitude and propulsion controls, abort switches and all functions requiring operation under acceleration are provided on the crew arm seats and require only finger tip operation. Two television cameras are provided for monitoring the crew and critical displays.

The drogue and main recovery parachutes are located in the forward section outside of the pressure cabin. The nose cap is jettisoned by the drogue mortar and is released automatically at a preset re-entry altitude, although a manual emergency deployment system is provided. Sun and star sensors and the VHF and C-band antenna are located in the nose cap. Three deployable doors, located around the periphery of the nose, house the fixed recovery system components, the recovery aids, the life raft, and survival equipment. Two of the pitch reaction control jets are mounted along the top center in a fixed section between two doors.

The aft equipment compartment contains the reaction control jets and propellants, spacecraft umbilicals and storage for lunar specimens. Four landing shock attenuators, dye marker, and shark repellant are located between the pressure cabin rear bulkhead and the heat shield support structure. The heat shield release can either be deployed automatically or by manual override.

The crew and equipment space allocations for the pressure cabin are shown on Figure 3-15. Space provisions for the crew members are based on data obtained from the NASA Manned Space Center, Houston, Texas for pressure suits inflated by 5 lb/in^2 pressure. The exit hatch



SPACE TECHNOLOGY LABORATORIES, INC.

OCT. 16, 1962 PD36-59

Figure 3-15 Pressure Cabin Crew and Equipment Space is also sized to provide clearance for a crew member with a shaped, back-pack life-support system used for extra-vehicular lunar operations.

The crew compartment equipment arrangement is shown in Figure 3-16. Crew convenience, accessibility for installation, checkout and servicing, minimum cabling, and inflight checkout were primary considerations in the location and arrangement of the equipment.

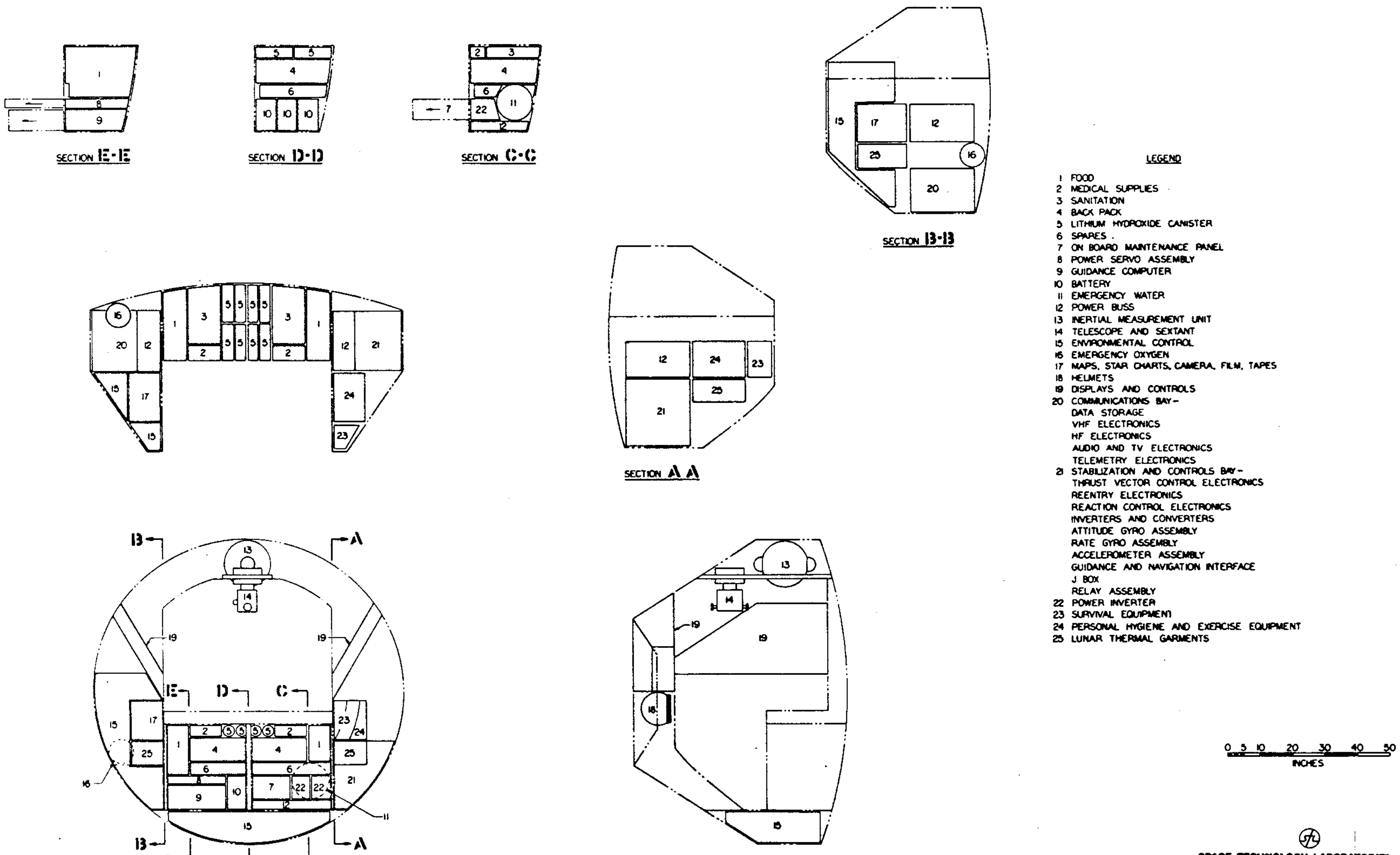


Figure 3-16 Equipment Arrangement Crew Compartment

SPACE TECHNOLOGY LABORATORIES INC.

OCT. 18, 1962 PD36-61

3.8.2 Subsystems

The present study included a detailed re-examination of all of the subsystems required for the support of the direct flight lunar mission. A detailed comparison was made between the subsystems used for the NAA Apollo and the 3-man system described in Reference 1. The results of the re-examination were used for the 2-man system definition. This subject is discussed in considerable detail in Volume II of this report where item by item design and weight comparisons are presented. Only brief descriptions of the 2-man direct flight subsystems are presented here.

3.8.2.1 Earth Landing and Recovery

The earth landing system uses a cluster of three Mercury project-type parachutes. A single drogue chute is contained over the main chutes under a jettisonable cap. Recovery aids, life rafts, and survival equipment are located around the periphery of the main parachute compartment. A comparison of the arrangements used in the present 2-man design and the earlier 3-man design is shown in Figure 3-17

For landing shock attenuation, four paper honeycomb pads are deployed (Figure 3-18). This configuration is suitable for landing with horizontal velocities up to 25 ft/sec and sink velocities up to 33 ft/sec.

3.8.2.2 Guidance and Navigation

The guidance and navigation system capability includes three major options which, taken altogether, provide a high degree of system redundancy. They are:

a) Redundant DSIF guidance equipment operating with the stellar up-dated MIT inertial guidance system.

b) MIT system with the capability for performing all guidance and navigation functions independent of earth-based information.

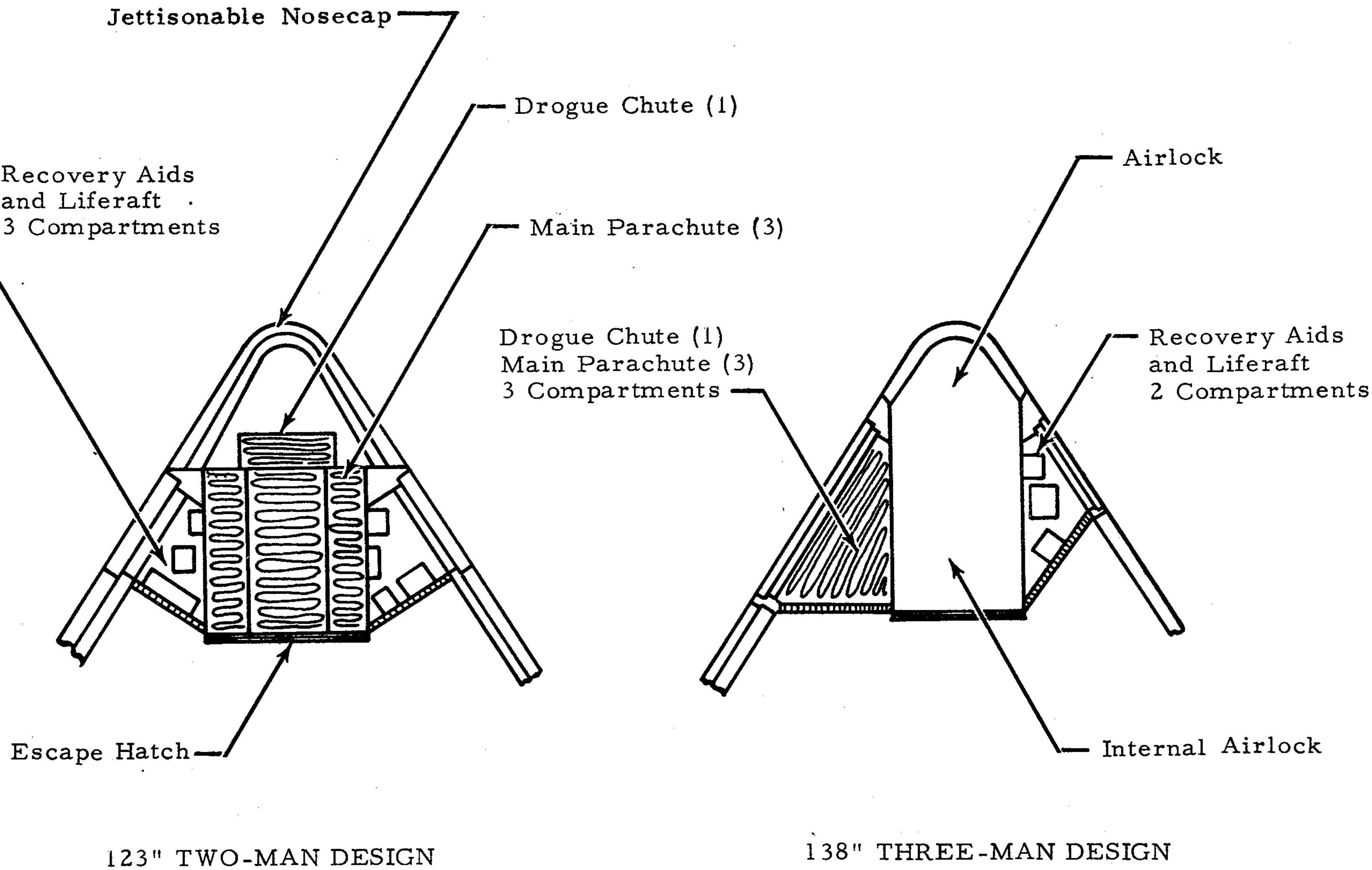


Figure 3-17 Recover System Arrangement for Two and Three-Man Command Modules

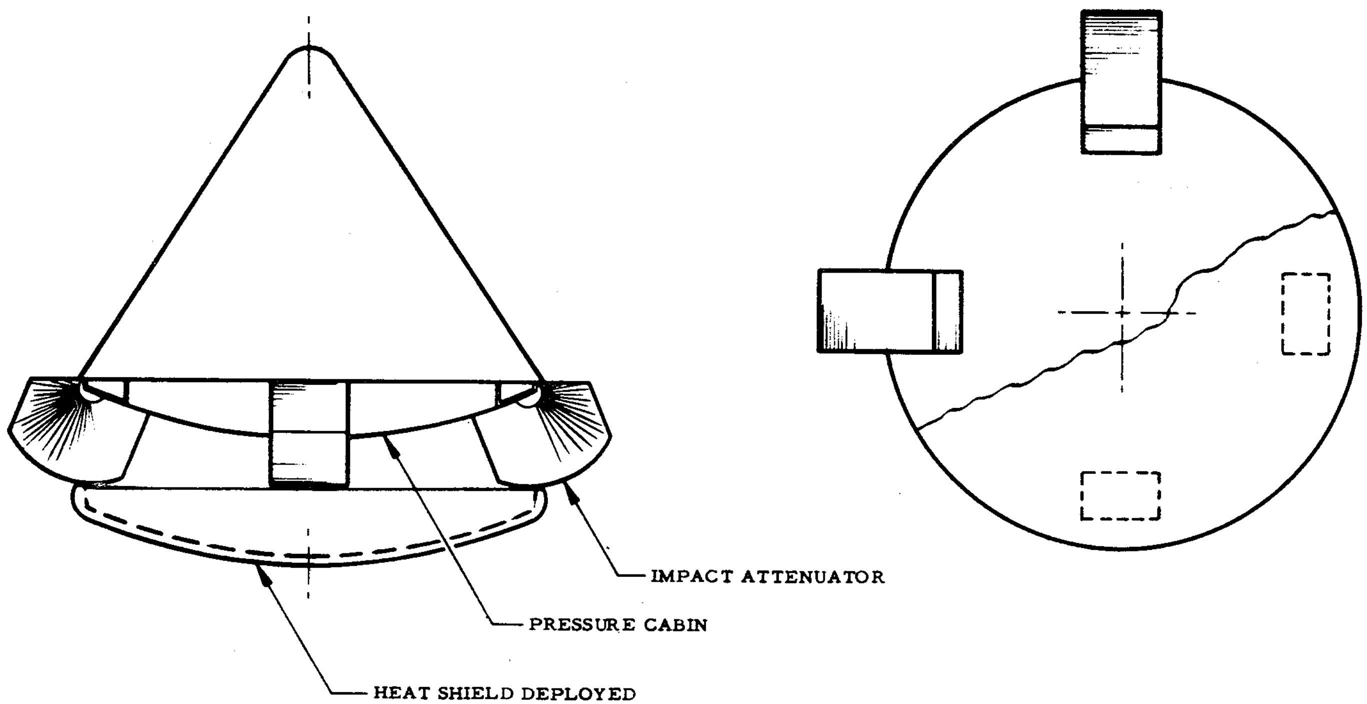


Figure 3-18 Earth Landing Impact Attenuators

SPACE TECHNOLOGY LABORATORIES, INC.
A SUBSIDIARY OF THOMPSON RAMO WOOLDRIDGE INC
ONE SPACE PARK • REDONDO BEACH, CALIFORNIA
OCT. 3, 1962 PD36-55

c) Redundant DSIF guidance equipment operating with a stellar up-dated stabilization and control system.

The equipment for these various modes can be operated in other combinations with varying degrees of crew participation. In addition, the DSIF modes can be operated fully automatically as desired. A block diagram of the guidance and navigation system is shown in Figure 3-19.

The equipment added to the Apollo design to provide these capabilities are: DSIF decoders which can read data directly into the MIT computer and a star tracker which works in conjunction with the sun tracker (similar to the one used in Apollo) to up-date the inertial attitude references.

The onboard autonomous (MIT) system requires more midcourse velocity and reorientation capability than does the DSIF stellar up-dated system. Thus, for example, on the flight to the moon, the MIT system requires about 50 reorientation maneuvers and 5 midcourse velocity corrections totalling about 300 ft/sec. The stellar updated DSIF system requires 2 reorientations and 2 midcourse corrections totalling about 112 ft/sec.

The DSIF and MIT systems provide the guidance and navigation capabilities needed for all flight phases except lunar landing. Doppler and altimeter radar systems and a landing TV system have therefore been included. With the system shown in Figure 3-19, the spacecraft has the capability to land either automatically or under the control of the crew using, in addition, lunar surface beacon aids if they exist. During the terminal descent, which might begin at an altitude of 50 to 100 feet, the crew could switch to pure inertial guidance to descend at the desired landing rate until touchdown. The altitude at which this mode might be initiated would depend upon lunar dust effects on the radar and television sensor operation.

3.8.2.3 Stabilization and Control

Inputs to the flight control system can come in the form of attitude commands from the guidance and navigation system, from the sun and star

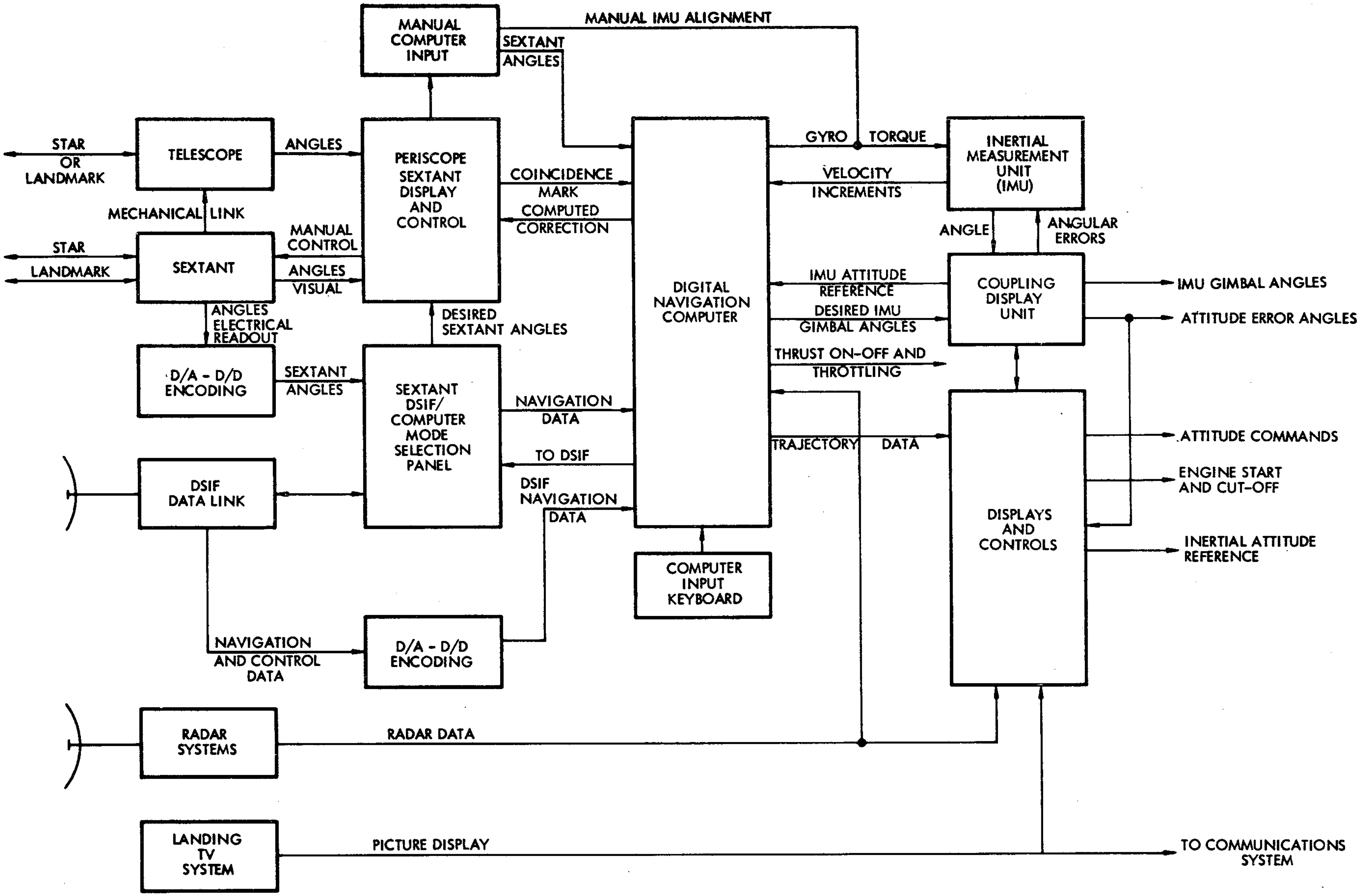


Figure 3-19 Guidance and Navigation Summary Block Diagram

sensor reference, and/or from the pilot through his manual controls. These signals are processed by the stabilization and control system electronics and sent as commands for thrust attitude deflections of the main engines or for pulse modulating the reaction jets.

Thrust vector control in pitch, yaw and roll is obtained during deboost stage main engine firing by gimballing the three RL10 engines. The maximum gimbal angle of 10 degrees is sufficient to handle the failure of any engine. Electrical actuators are used for gimballing to simplify hardware mechanization and avoid problems from the low temperature environment.

During power-on operation of the landing and takeoff stage, attitude control is obtained by translating the main engine. Actuator response is designed to be slow since only a trim capability is needed to takeout center of gravity offsets. For fast response requirements, a system of 100 pound thrust reaction jets are used.

These jets are also used for vernier midcourse velocity corrections during the coast phases and for attitude orientation and stabilization during the earth-to-moon transit. On the return leg, the spacecraft mass properties are sufficiently reduced so that the minimum angular velocities produced by the 100 pound jets exceed the low attitude rates specified for the MIT guidance system. A low thrust level system of jets, 5 pound thrust, are, therefore, used during the moon-to-earth coast. The low level thrust system also provides substantial savings in propellant consumption during moon-to-earth limit cycling.

A block diagram of the stabilization and control system is shown on Figure 3-20. Redundant control electronics are provided for reaction and thrust vector control, rate gyro assemblies and power converters and inverters. In addition, completely redundant systems of reaction jets are provided on the lunar landing and takeoff stage and command module.

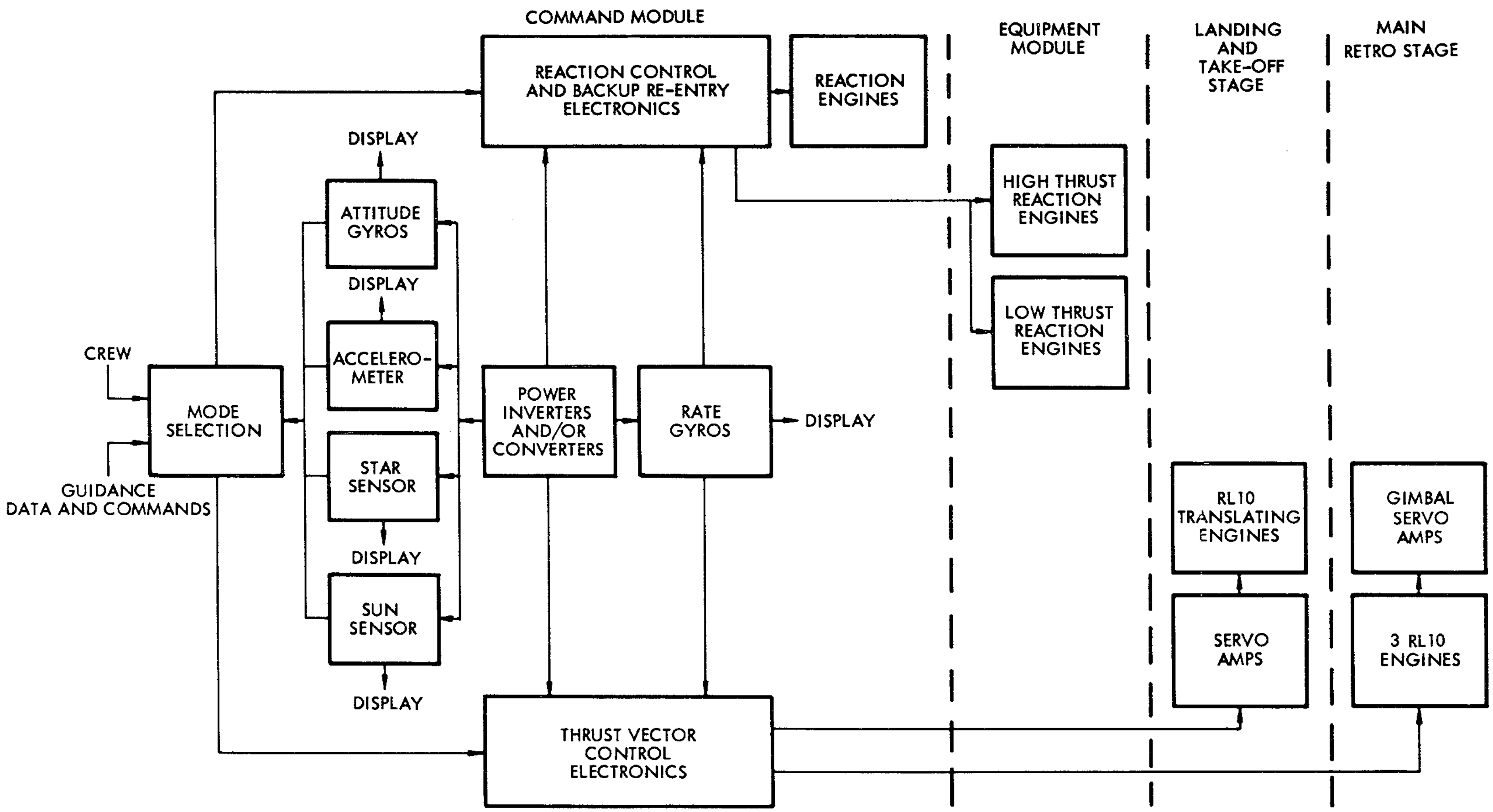


Figure 3-20 Stabilization and Control System Block Diagram

The stabilization and control system also provides a backup re-entry guidance capability. This is provided with the longitudinal accelerometer, roll attitude gyro, nominal drag acceleration and roll rate programmers and associated electronics. Analog computer simulations of super orbital velocity entry were made with this system. They show that the guidance system operates satisfactorily when re-entering within 0.1 degree of the safe corridor. Downrange miss is of the order of 40 nautical miles for an 1800 nautical mile nominal range with a one degree (greater than 3σ after 2 DSIF corrections) re-entry error.

3.8.2.4 Reaction Control

The reaction control systems on the command module and lunar landing and takeoff stage use earth storable hyperbolic propellants (N_2O_4 and N_2H_4 - UDMH, 50-50). The systems consist of helium pressurization tanks, positive expulsion propellant tanks and redundant pairs of reaction jets in each control channel. There are, therefore, 6 pairs of 100 pound thrust reaction jets on the command module and 6 pairs each of 100 pound and 5 pound thrust jets on the lunar takeoff and landing stage.

The thrust units on the command module are ablative cooled and buried within the structure. A chamber pressure of the order of 150 psia is selected to minimize engine size and weight. For the landing and takeoff stage, radiative cooled engines operating at a chamber pressure of 60 psia are used.

3.8.2.5 Communications and Instrumentation

The communication and instrumentation system provides a capability for near earth HF and VHF communications and instrumentation, and C-Band and minitrack beacon tracking. For distances beyond a few thousand miles, all communications, instrumentation and tracking are conducted using the Deep Space Instrumentation facility. Data capability is provided for voice, television, telemetry, on-board data storage and play back using tape recorders. The personnel communication system permits voice communication between crew members outside or inside the command module either with each other or direct to earth.

Using a 10-watt rf transmitter the signal-to-noise margins are adequate for use with the 85-foot diameter earth-based antennas. The airborne system is compatible for use without modification with the 210-foot diameter antennas with an increase in signal-to-noise of 8 db.

The spacecraft uses 2 storable parabolic 3-foot diameter antennas and an omnidirectional DSIF antenna. Figure 3-21 shows the block diagram of this system.

3. 8. 2. 6 Crew and Crew Support

The crew and crew support subsystem includes the crew members, their garments, pressure suits, back packs for extra vehicular operation, seats, restraints, food, emergency water, biomedical equipment and supplies, sanitation equipment, and atmospheric control and survival equipment. The system has been sized for 90^{th} percentile men.

The atmospheric supply is designed to permit 18 airlock operations and 2 cabin repressurizations. The spacecraft cabin is maintained at a nominal pressure of 7 psia. The cabin atmosphere is composed of oxygen at a partial pressure of 3. 5 psia, nitrogen at 3. 25 psia, water vapor to provide a relative humidity of 50 percent and a small amount of carbon dioxide. In the event of an emergency decompression, the pressure suits will inflate with pure oxygen to a pressure of 3. 5 psia with provisions for increase to 5 psia if needed. The atmospheric gases are stored as supercritical fluids in the lunar landing and takeoff stage except for the small re-entry supply which is stored as high pressure gas in the command module.

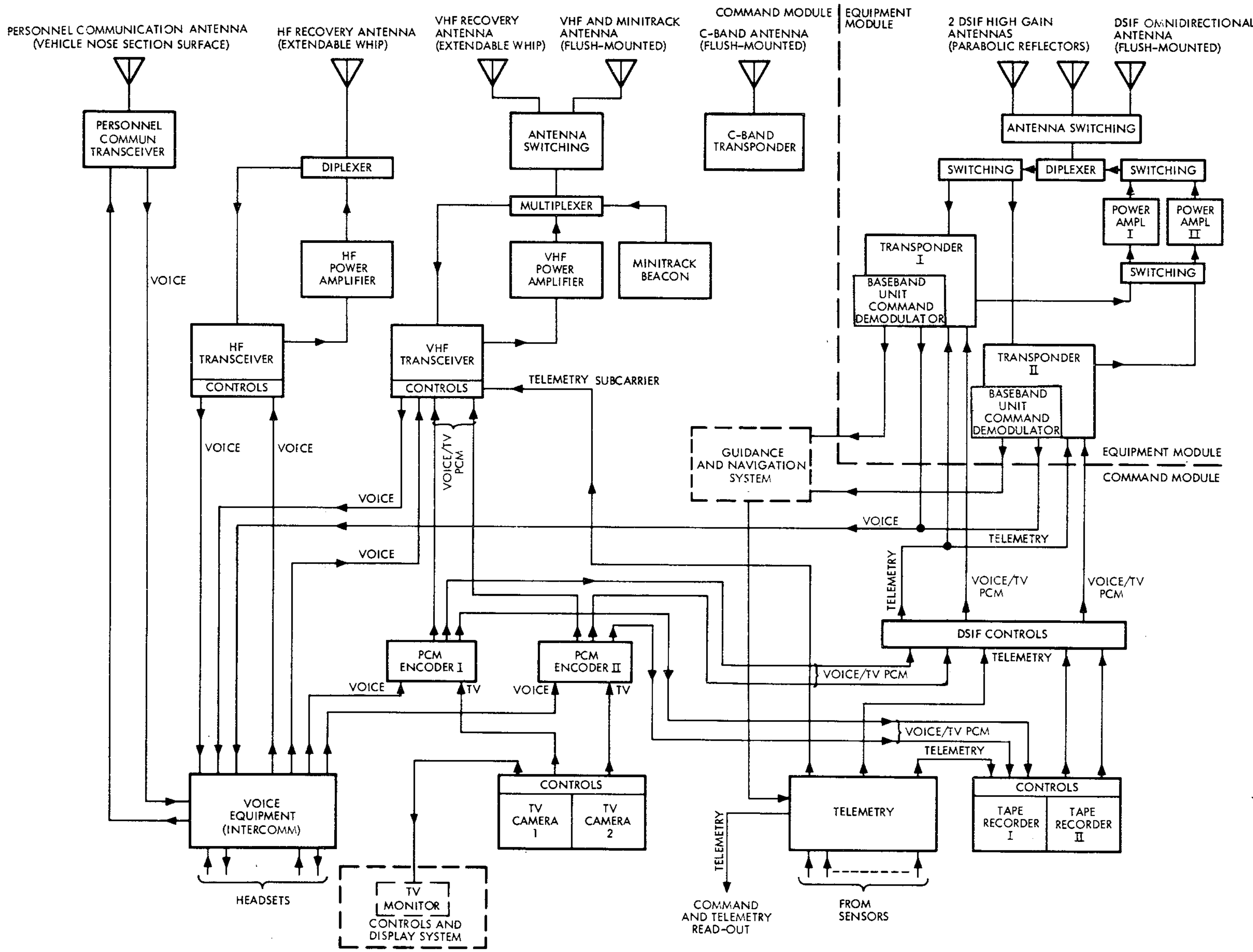


Figure 3-21 Communications and Instrumentation Block Diagram

[redacted]

The crew support and restraint system used in the present study is a result of STL company-funded investigations. The system is less bulky than the full contoured couch and provides more nearly optimum restraint in all directions than the contoured couch used in Project Mercury. Figure 3-22 shows the restraint system. The primary shock attenuation is provided by the crushable honeycomb structures deployed between the heat shield and the command module. The suits and restraint equipment provide a small additional amount of shock protection and some attenuation is also provided in the seats by crushable plastic foam which lies between the structure and the contoured seat surface panels.

Food is furnished primarily as a semi-fluid or paste-like preparation and packaged in soft collapsible plastic tubes. The food while of uniform composition can be flavored to closely resemble a number of preferred foods. The dietary regimen supplies 2300 kilocalories, 115 grams of protein, 77 grams of fat and 288 grams of carbohydrate per day. The primary water supply is obtained from the fuel cell power supply and from the condensate produced in the water separator of the environmental control system. Thirty-six pounds of water are stored in the command module for emergency survival and re-entry environmental cooling.

3.8.2.7 <u>Environmental Control</u>

The environmental control system provides thermal control for the equipment and crew. The system is closely integrated with the electrical power supply system and the crew support system. A schematic diagram of the system displaying the interrelationships is shown in Figure 3-23. The system definition differs from that used by NAA in the Apollo where the carbon dioxide removal units, detoxifiers, atmosphere control unit, water tanks and pressure suits are considered to be part of the environmental control system.

[redacted]

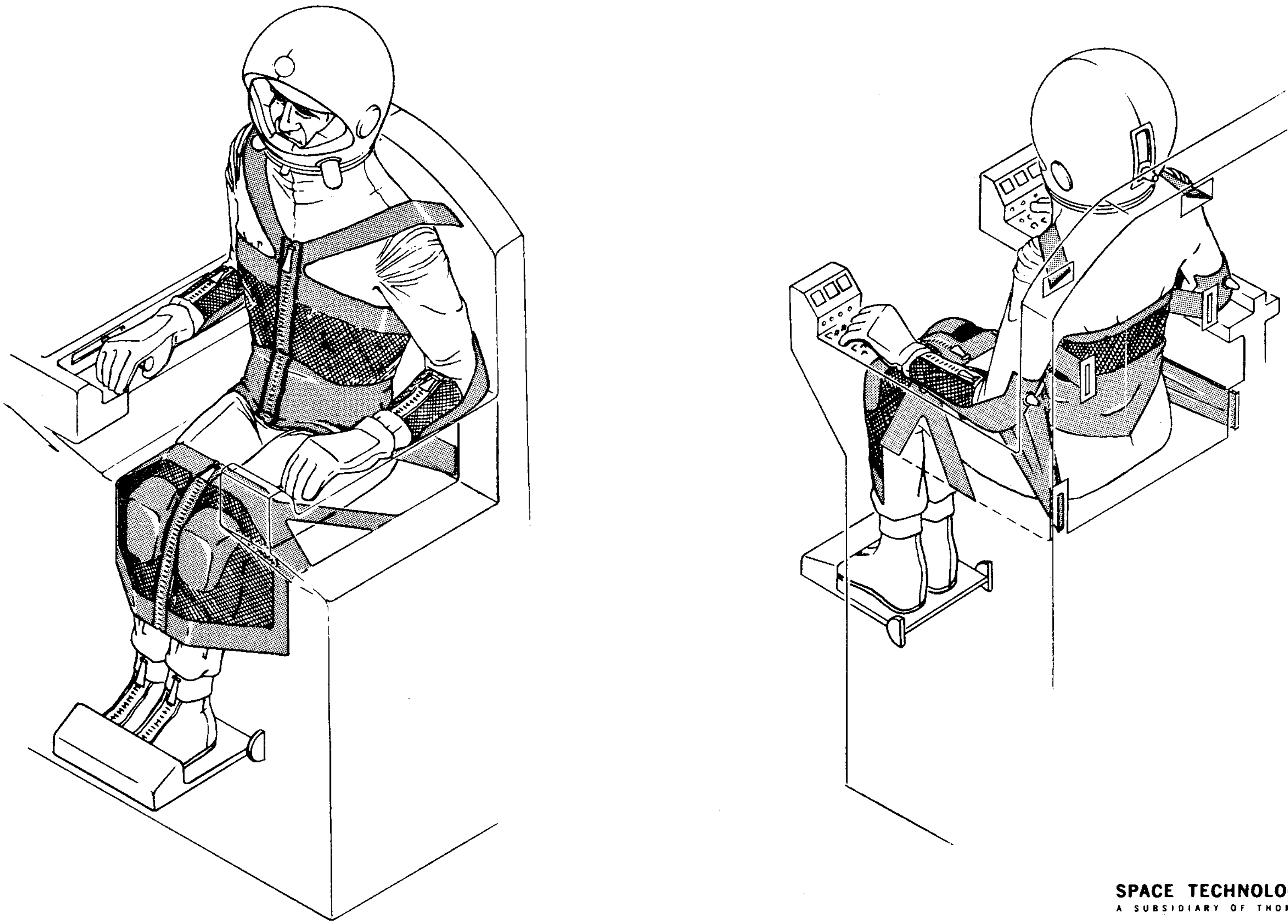


Figure 3-22 Crew Restraint System

[redacted]

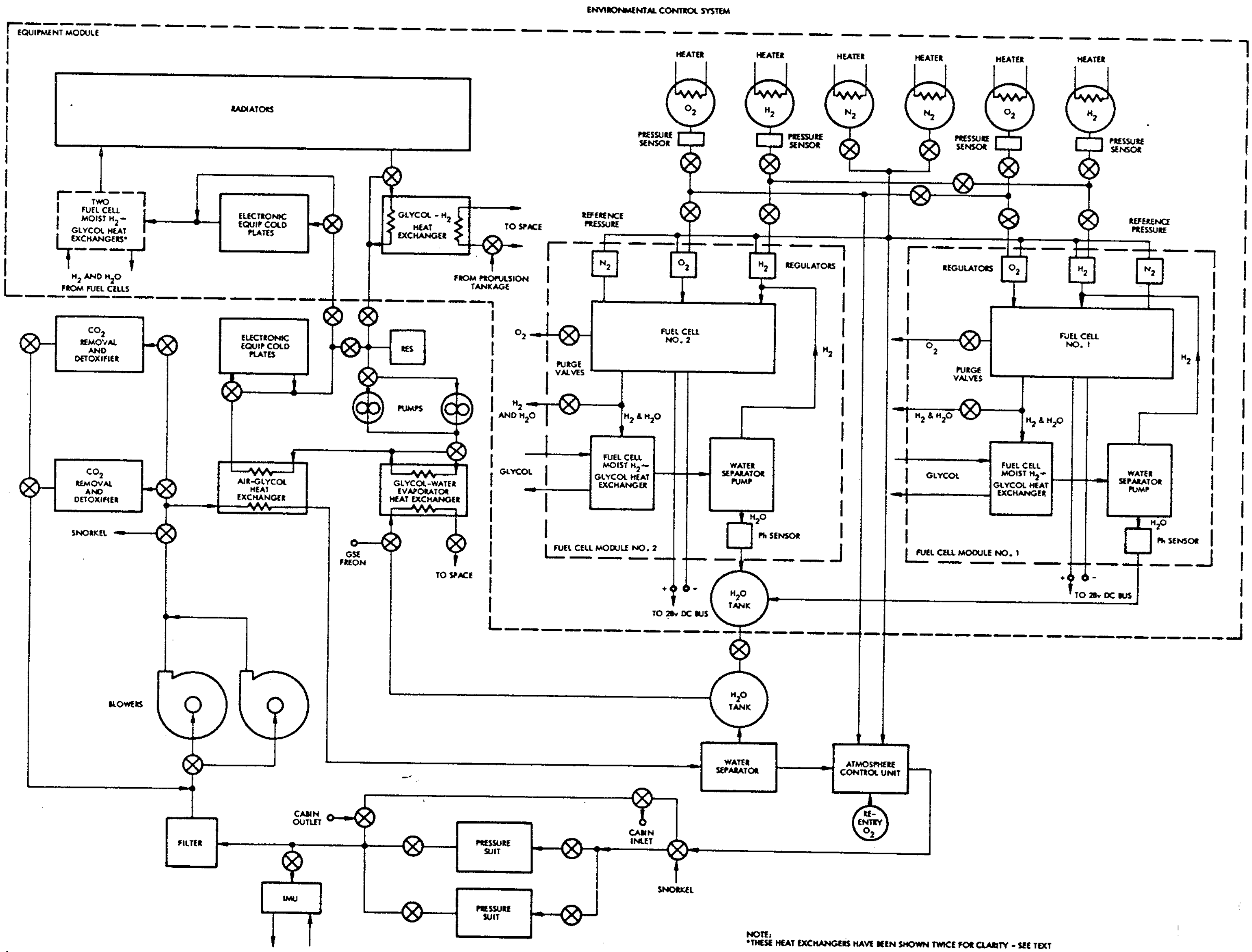


Figure 3-23 Schematic Diagram of Environmental Control System

CO[redacted]

The air leaving the pressure suits and cabin is drawn through the filter to remove particles in the air. Prior to entering the filter, some air is tapped off and circulated inside the IMU for cooling. The air then passes through a centrifugal blower. Approximately 10 percent of the flow circulates through the carbon dioxide removal units and detoxifiers and returns to the inlet of the blower. After leaving the blower, the air passes through the air-glycol heat exchanger and on to the water separator where excess moisture and water droplets are removed.

In the water-glycol loop, the glycol passes directly from the pump through the air-glycol heat exchanger. The glycol passes through the air-glycol heat exchanger removing the metabolic heat of the crew, the heat dissipated by the electrical equipment, and any heat leaking into the cabin. This total heat load could be as high as 579 watts. During re-entry (or during emergency), the glycol is routed through the glycol-water evaporator heat exchanger after leaving the pump to make use of water cooling.

The glycol then passes through the IMU and electronic equipment cold plates in the command module and then into the equipment module and through the remaining electronic equipment cold plates. It then passes through the fuel cell moist hydrogen-glycol heat exchangers. In transit the radiators are deployed and rotated 180 degrees. The glycol leaves the radiators at $40^{\circ}F$.

On the lunar surface during daytime conditions the radiators are deployed parallel to the surface for maximum effectiveness. During the hottest portion of the lunar day, the radiator cooling is augmented by a glycol-hydrogen heat exchanger which uses hydrogen boiloff from the lunar takeoff stage propellant tanks. During the lunar night, the radiators can be in a closed position with the insulation on the outer surface. However, some portion of the radiators would remain exposed since the internal heat generated is greater than the heat leaks by conduction to the ground or radiation to space from the vehicle surfaces.

3.8.2.8 Electrical Power System

The source of spacecraft power consists of two 1 kilowatt hydrogen-oxygen fuel cells operating in parallel. The units chosen are the Bacon-type built by Pratt and Whitney and are based on Apollo technology. The size of the fuel cell is reduced slightly from that used by Apollo in that 27 cells are connected in series, as suggested by Pratt and Whitney, instead of 31. It is estimated that under emergency conditions, either fuel cell can supply as much as 1100 watts at 27 vdc (measured at the fuel cell).

During normal operation, each fuel cell supplies one-half the total electrical load, but in the event of failure of one of the cells, the other could supply the total load without causing battery drain except during periods of peak power. If the total load during peak power periods was reduced by approximately 400 watts, a single fuel cell could still supply the total load without causing battery drain at the lower specification limit of system voltage. This can be accomplished by turning off the onboard guidance equipment, and relying on DSIF guidance (which is 100 percent redundant).

The multiple hydrogen and oxygen tanks are connected such that either or both fuel cells can be supplied by any combination of hydrogen and oxygen tanks. The hydrogen and oxygen tanks contain heaters and pressure sensors. A control system will sense tank pressure and supply heat to maintain the initial pressure in the tanks as the reactants are used and until the tanks are at ambient temperature. At this time, the pressure will be allowed to drop until it reaches 60 psi, at which time all of the reactants will have been consumed except for a residual calculated to be less than 1 percent. Nitrogen gas is supplied to the fuel cells from the atmospheric control unit, but it is used only as a reference pressure and no nitrogen gas is consumed by the fuel cells.

The average power load for the entire mission is 1018 watts and the peak power, which occurs just prior to lunar landing, is 1578 watts. The total power produced by the fuel cells during the entire 8-day mission (not including re-entry and recovery) is 195.5 kilowatt-hours. Figure 3-24 shows the peak periods of the power profile on an expanded time scale.

A schematic of the electrical system is given in Figure 3-25. It shows the two cells connected to redundant equipment module buses which are in turn connected to redundant primary buses within the command module. All the buses include cross-tie contactors to provide for system flexibility. Essential loads are supplied from the equipment module and primary buses respectively depending on their location. The system includes a nonessential bus which can be fed from either primary bus through circuit breakers, and which can be disconnected either automatically or manually in case of an emergency.

Three low-rate, sealed silver-zinc batteries are used to furnish electrical power during earth re-entry, landing, and post landing. The three batteries (two re-entry and one post landing) are equal in size and have a total capacity of 4700 watt-hours. Either of the two redundant re-entry batteries can supply the total re-entry load for 1.6 hours thus providing considerable margin.

Inverters provide 115 volt 400-cycle ac power to the various ac motors in the system. These include glycol pumps, environmental control air blowers, fuel cell water separator pumps, and antenna drive motors. The centralized inverters provide only motor power since these units can use square wave, unregulated power, thus making it possible to use a high efficiency inverter for a relatively high ac load (approximately 325 watts). All other ac loads are supplied by inverters within the individual subsystems to meet particular requirements for wave shape, regulation, frequency control, etc.

Two redundant ac buses are provided; either bus can be powered by either inverter which in turn can be fed by either primary bus. Ordinarily, one inverter is connected to both primary buses and both ac buses while the

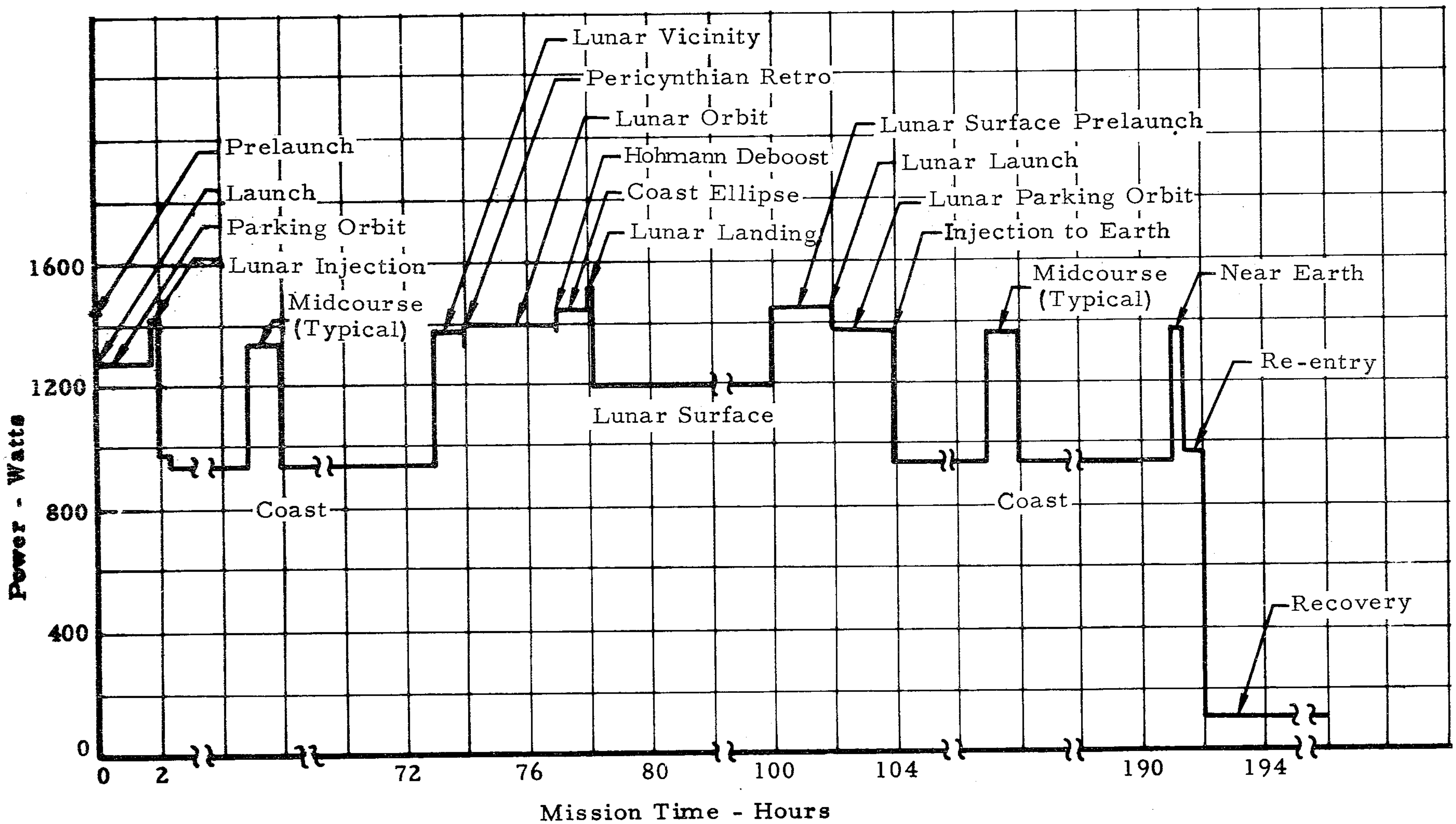


Figure 3-24 Expanded Scale Power Profile

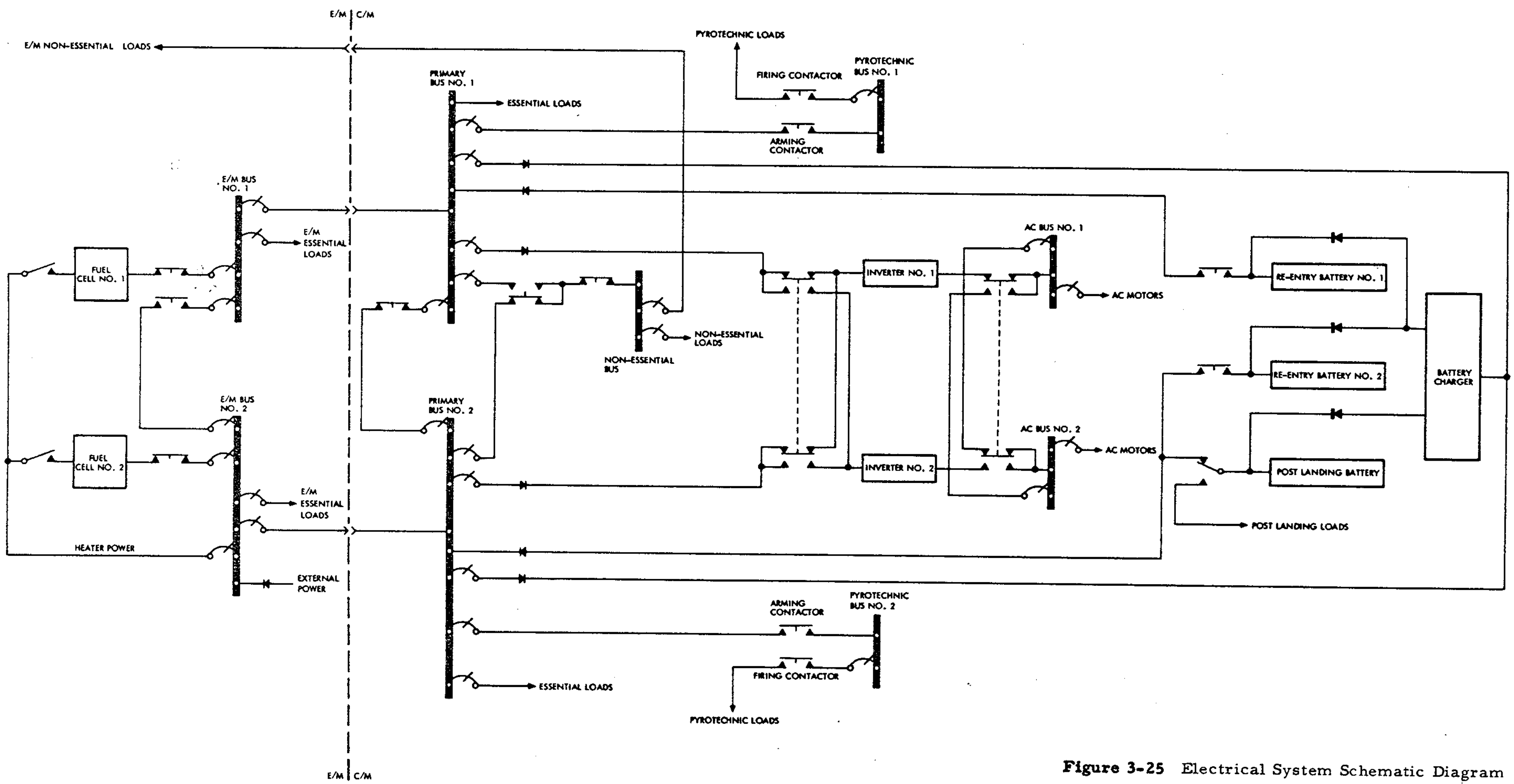


Figure 3-25 Electrical System Schematic Diagram

other inverter is held in redundant standby. Pyrotechnic loads are supplied through redundant pyrotechnic buses which are isolated from the primary buses by circuit breakers and arming contactors.

3.8.2.9 Control Panels and Displays

Figure 3-26 shows the control and display arrangement. The pilot and copilot basic flight and critical display panels are located in a 30-degree vision cone forward of the crew approximately 36 inches from the eyes when seated. These displays provide visual data only and do not require the crew to operate controls on the panel itself during high acceleration maneuvers.

Additional displays and controls are located on the sloping console panels at the side of each crew member. Each console consists of a fore and aft panel. Controls and monitors are arranged for easy access during all conditions of acceleration.

3.9 RESCUE MODE

The rescue mode defined for the present study required unmanned lunar landing and unattended lunar stay up to at least 30 days. The modifications of the basic system to accomplish this mode are minimal except possibly from the micrometeoroid hazard standpoint.

The guidance and navigation system described by STL permits automatic lunar landing using a programmed sequence of events. Landing aids in the form of beacons can permit a more precise control of the terminal landing dispersion. Alternatively, earth controlled lunar landing can be accomplished using television aids and the DSIF system. These modes have been extensively investigated under other NASA contracts (Reference 8).

The lunar takeoff stage propulsion system requires somewhat more insulation for a protracted stay on the moon. However, either earth storable or cryogenic propellants are feasible. For the cryogenic system, both the insulation and boiloff are increased by a factor of about

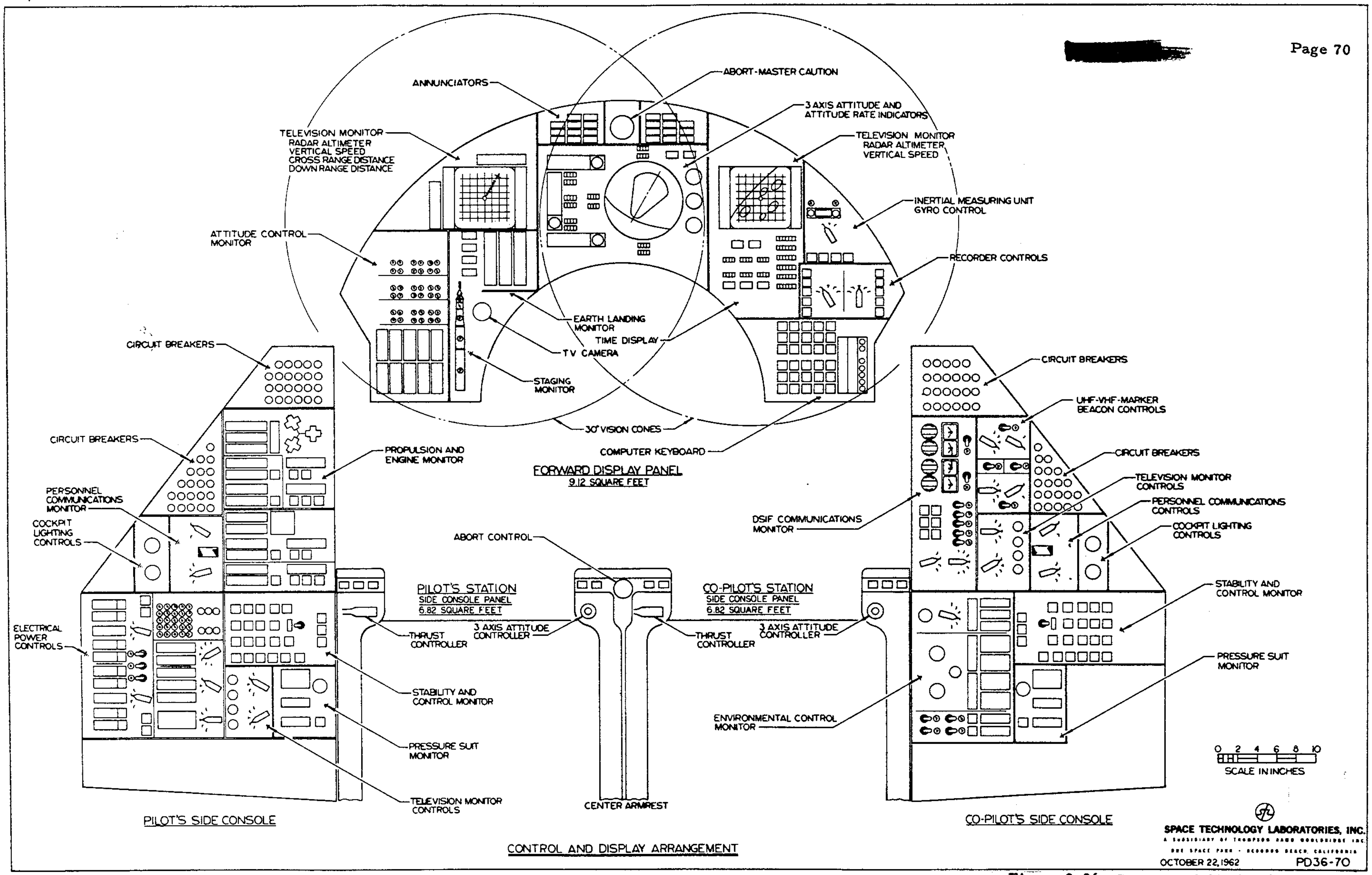


Figure 3-26 Control and Display Arrangement

[redacted]

[redacted]

2 for a 30-day period compared to a normal mission. This would reduce the allowable payload by about 500 pounds which is well within the system margins.

The communications and instrumentation system requires the addition of a few pounds of data handling and sequencing equipment to permit commanding on and off DSIF, telemetry and beacon systems. An X-band transponder should also be added to operate as a beacon to mark the location of the rescue vehicle for the orbiting and landing manned spacecraft.

The environmental control system may require the addition of a small heating system (electrical elements or hydrogen-oxygen gas burner) in the compartment beneath the command module to keep the heat shield temperature up to about 0° F during the lunar night. Temperature control of the equipment in the command module can be accomplished with the basic environmental control system and a small continuous expenditure of power.

The fuel cells in the electrical system can be operated in an "idling" mode. Under these conditions, the two cells will produce about 100 watts of electrical power and 140 watts of heat. For a 30-day period, this will require an additional 180 pounds of hydrogen, oxygen and tankage. The average power produced is greater than required.

The present uncertainty in the micrometeoroid hazard makes an estimate of required penetration protection quite conjectural. The probability of penetration of a vital component of the STL vehicle during a normal 8-day mission may be somewhere between once every 3 missions and once every 1000 missions. The effective thicknesses required for pressure vessels and other vital components cannot therefore be specified. It is, however, possible to compute the additional structural weight required to make the hazard of a 30-day mission no greater than the hazard of an 8-day mission regardless of the magnitude of the hazard. This results from the form of

[redacted]

[redacted]

the equations used which states that the number of penetrations per day varies inversely with the cube of the skin thickness. Based on the material gauges used, one can compute that the command module weight should be increased by about 75 to 80 pounds and the lunar takeoff stage by about 565 pounds. The weight increases are (at least partially) offset by a weight saving on the earth-to-moon leg. It is estimated that at least 380 pounds (including the crew) are removed. It is concluded that the rescue mode is feasible with the two man system and does not require too extensive modification. The incremental weight requirements are within the performance capabilities of the system.

3.10 COMPARISON OF THE STL 3-MAN SYSTEM DESIGN OF REFERENCE 1 WITH THE NAA APOLLO

An extensive and detailed comparison was made between the STL 3-man design of Reference 1 and the NAA Apollo. The comparison covered design criteria and its relation to the NASA Apollo guidelines as well as the differences in subsystem mechanization and weights.

The principle divergences from the NASA guidelines were in the following areas:

a) The use of a smaller command module diameter and a shorter mission duration than specified.

b) Different arrangement of the command module and service module.

c) The use of pump fed cryogenic propellants for lunar spacecraft propulsion.

d) Reduction of electrical power requirements and deletion of one fuel cell.

e) Elimination of personnel parachutes, separate escape hatches, privacy curtains and other niceties.

f) Shock mitigation was provided for the entire capsule rather than just for the crew.

[redacted]

[redacted]

There were also many minor divergences from the NASA guidelines. However, it was interesting to note during the study that the NAA system is also diverging from the guidelines as the pressure for weight reduction forces a reconsideration of criteria and system mechanization.

The problem of reconciling the weights used by STL in its subsystem design with those published by NAA proved to be quite difficult for several reasons. For one, the subsystem definitions used by the two companies were different so that the allocation of component weights could not easily be made. Second, the published specifications to which NAA were sizing or designing equipment were incomplete. It is believed, however, that most of the equipment weights were correlated and that only a few unreconciled areas remain. Detailed comparisons of subsystem weights are presented in Volume II; a brief summary of some of the major differences are presented below:

a) Structure and heat shield weights are largely size dependent. The external surface area of the NAA 154-inch command module is about 25 percent larger than the STL 138-inch command module. Many of the internal areas have proportionately larger ratios because specific dimensional constraints do not permit linear scaling internally. In addition, some of the structural weight is dependent on recovery and landed weight so that the basically heavier vehicle is penalized by the so-called growth factor. The weight of the original 138-inch command module structure and heat shield is in good agreement with the NAA weights. However, the weights of these subsystems is much greater than needed and a much more efficient (and lighter) design was found for the 123-inch diameter command module studied in detail under the present contract.

b) In the STL system, there is a considerable integration of electrical, environmental and crew support subsystems. Some of the weight differences result from this factor while others are attributable to differences in system and mission requirements. A few of these are listed below.

[redacted]

[redacted]

1) The reduction in average power level and the use of 2 rather than 3 fuel cells saves about 380 pounds in fuel cell weight.

2) The reduction in mission duration from 14 to 10 days and the reduction in power level saves about 420 pounds in hydrogen, oxygen and tankage.

3) NAA includes 176 pounds of main propellant tank insulation under the environmental control system while STL includes this insulation in the propulsion stage weight.

4) The longer mission time used by NAA also accounts for a requirement for more oxygen and food for crew support.

c) The earth landing and recovery system weight shown by NAA is about 230 pounds heavier than the STL system. About half of this is attributable to the heavier weight of the NAA capsule at the time of parachute deployment. The remaining difference results from the lower horizontal wind criteria used by STL in the design of the shock attenuation system.

d) As an illustration of the effect of size, mission duration, number of crew and the general growth factor, Figure 3-27 presents a summary parametric weight curve for the command module and associated support equipment.

[redacted]

[redacted]

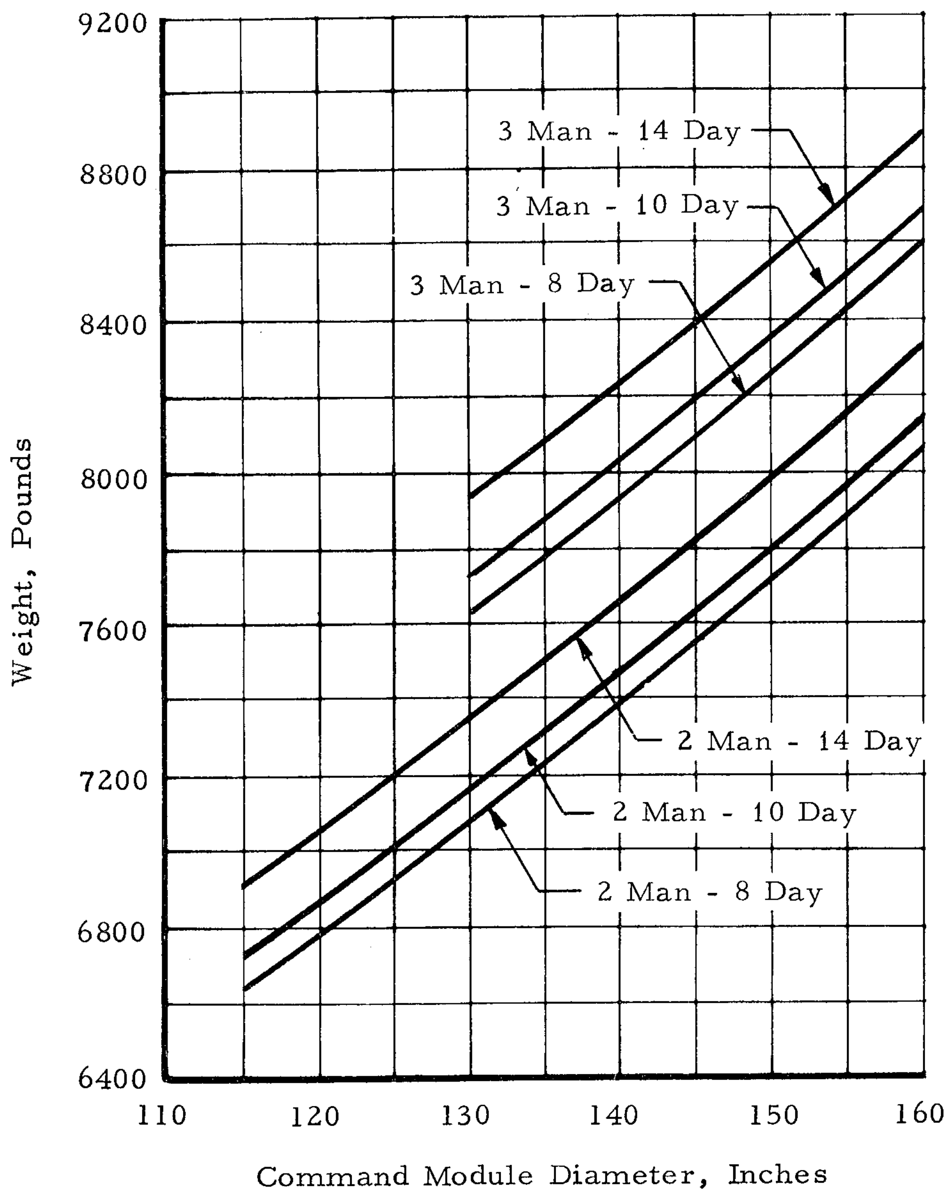


Figure 3-27 Command Module and Support Equipment Gross Weight.

[redacted]

4.0 REFERENCES

(1) STL Document No. 8667-6014-TC0000, "Direct Flight Schedule and Feasibility Study for Project Apollo," 25 June 1962, Contract No. NASw-443 (C)

(2) NASA, "Project Apollo Spacecraft Development Statement of Work Part III--Technical Approach," dated 18 December 1961, Manned Spacecraft Center, Langley AFB, Virginia (C)

(3) NASA Memo, M-MS 1411.205 MSA (MS:dk) MSC, "ΔV Requirements Presently Being Used by MSC in Their Service Module and LEM Activity," from M. Savage to E. Hall, Dated 2 August 1962

(4) NASA Memo, "Mission ΔV's" from M. E. Alper to A. Fiul, dated 22 August 1962

(5) NASA Memo, "Final Resolution of ΔV's for the Two-Man Direct Studies," by M. E. Alper, dated 18 September 1962

(6) NASA, Ames Research Center, Moffett Field, California, Paper 2644-62 "Minimum Crew Space Habitability for the Lunar Mission," by G. A. Rathert, Jr., N. M. McFadden, R. F. Weick, R. M. Patton, G. W. Stennett, and T. A. Rogers. Paper presented at the ARS 17th Annual Meeting, 13 November 1962

(7) STL Document No. 9862.1-51, "Two-man Command Module for the Direct Lunar Mission for NASA Office of Manned Space Flight," by J. T. Bennett, F. R. Crandall, and D. R. Pence, dated 24 July 1962

(8) STL Document No. 9990-6022-RL-000, "Analysis of Systems for Accurate Lunar Landings," dated 4 January 1962 (C)